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STUDY FOR ANALYSIS OF BENEFIT VERSUS COST OF LOW THRUST PROPULSION SYSTEMS

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16 Abstract A study was conducted to (1) investigate the benefits and costs associated with placing Large Space Systems (LSS) in operational orbits and (2) develop a flexible computer model for analyzing these benefits and costs. A mission model for LSS thru the year 2010 was identified that included both NASA/Commercial and DOD missions. This model included a total of 68 STS launches for the NASA/Commercial missions and 202 launches for the DOD missions. The mission catalog was of sufficient depth to define the structure type, mass and acceleration limits of each LSS. Conceptual primary propulsion stages (PPS) designs for orbital transfer were developed for three low thrust LO_2/LH_2 engines baselined for the study. The performance characteristics for each of these PPS was compared to the LSS mission catalog to create a mission capture.

The costs involved in placing the LSS in their operational orbits were identified. The two primary costs were that of the PPS and of the STS launch. The cost of the LSS was not included as it is not a function of the PPS performance. The basic relationships and algorithms that could be used to describe the costs were established. The benefit criteria for the mission model were also defined. These included mission capture, reliability, technical risk, development time, and growth potential. Rating guidelines were established for each parameter. For flexibility, each parameter is assigned a weighting factor.

The benefit and cost relationships were then programmed into a computer model that determines the benefit rating and cost of each engine. A sample problem was evaluated using the model to compare the three engines baselined for the study. To show the flexibility of the program several additional problems were examined. These included evaluating the impact of improving LSS structural strength and modification of the mission model to permit 100% mission capture by all three baselined engines.

This study developed a flexible computer model for evaluating the benefits and costs for launching and orbit transfer of any mission catalog. The model at present contains the performance envelopes of three primary propulsion systems for orbit transfer based on three low thrust engines baselined in the statement of work. However, it is possible to modify the basic model to examine any propulsion system. The model also allows for any mission model to be input into the program. The model also allows the user to easily vary the program to examine the effects of various ratings and weighting of benefit parameters for the baseline engines.

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LIST OF SYMBOLS USED IN MAIN TEXT

ACS	Auxiliary Control System
AFRPL	Air Force Rocket Propulsion Lab
ASDS	Advanced Spacecraft Deployment System Study
ASE	Airborne Support Equipment
CER	Cost Estimating Relationship
DLTE	Dedicated Low Thrust Engine
DOD	Department of Defense
ECAPS-LSS	Study of Electrical and Chemical Propulsion Systems for Auxiliary Propulsion of Large Space Systems
GEO	Geosynchronous Earth Orbit
LCC	Life Cycle Cost
LeRC	Lewis Research center
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
LSS	Large Space System
LTPS	Low Thrust Chemical Orbit to Orbit Propulsion System Propellant Management Study
MMDA	Martin Marietta Denver Aerospace
NPSH	Net Positive Suction Head
PP/LSSI	Primary Propulsion/Large Space System Interaction Study
PPS	Primary Propulsion System
RACE	Rating and Cost of Engine
RDT&E	Research, Development, Testing, and Engineering
STS	Space Transportation System
T/M_f	Final Thrust to Mass or Acceleration at Burnout

NASA is currently developing plans for missions requiring the utilization of Large Space Systems (LSS). These LSS's will be carried to Low Earth Orbit (LEO) by the Space Transportation System (STS-Shuttle). The predominant mission scenario is that these systems will be erected and/or assembled in LEO, and then transferred to Geosynchronous Orbit (GEO). Due to loading constraints, current chemical propulsion systems may not have the capabilities to meet the requirements of many of these LSS missions.

The NASA Lewis Research Center (LeRC) has been supporting efforts on various low thrust chemical propulsion concepts which can meet the LSS requirements. In order to assess the economic justification for these concepts, a study was initiated to quantify the benefits/costs.

The performance of three LO_2/LH_2 engine concepts, specified by LeRC, were compared with the propulsion requirements of NASA/Commercial and DOD LSS mission models to quantify the benefits and costs.

The three engine concepts specified for this study were:

(1) A dedicate low thrust engine with a thrust range of 890N (200 lb_f) to 4480N (1000 lb_f).

(2) An advanced engine with a thrust range of 4480n (1000 lb_f) to 66,700N (15,000 lb_f).

(3) An updated RL-10 engine with a thrust range of 6670N (1500 lb_f) to 66,700N (15,000 lb_f).

A scenario of 202 STS launches comprising the time frame up to 2010 was developed during the study for deployable LSS to be operated in GEO. Missions included only LSS's with deployable dimensions of over 30 m that could be transferred from LEO to GEO by the Primary Propulsion System (PPS).

Spacecraft that were too large for a single shuttle flight were only considered if they could be split into 2 or 3 launches for NASA/Commercial missions or up to 6 STS Launches for the DOD missions with each section flown separately to GEO.

A benefits and cost model was developed to compare Primary Propulsion Systems (PPS). The benefit algorithm is based on a weighted criteria rating approach. Benefit criteria selected are mission capture, reliability, technical risk, growth potential, length of development, technical desirability, stage length, system fabricability, and repairability in orbit. The cost algorithm defines the LCC as the payload deployment system from earth to final orbit. RDT&E costs and first unit costs are derived for various propulsion subsystems and summed to yield PPS values. Combination of these two algorithm resulted in a benefit and cost model which iterates on thrust level such that the most cost effective and beneficial thrust level is selected for a given mission catalog.

A sample comparison based on benefits/costs of the three LO_2/LH_2 engine determined that dedicated low-thrust PPS is the best system for both mission catalogs. The optimum thrust level for this PPS is 3400-4450 N (760-1000 lb_f). LCC of the dedicated low-thrust PPS to capture the total NASA/Commercial Mission Catalog is $4.6 Billion.

It is recommended that the benefits versus costs of relaxing the upper thrust limit of the dedicated low-thrust primary propulsion system be investigated.

I. INTRODUCTION

With the advent of an operational space transportation system (STS), NASA will have the capability of transporting large-volume/low-density payloads to low Earth orbit (LEO). Some of these will be structures that allow placement of very large antennas (> 200 m diameter), or collections of communication systems, in orbits ranging from LEO to geosynchronous Earth orbit (GEO). Currently one approach is to deploy these large space systems (LSSs) in LEO and transfer them to their operational orbit by a primary propulsion system (PPS). The vehicle thrust must be limited to assure loading during the final acceleration will not collaspe the lightweight structure.

The objective of this program was to investigate and model the benefits/cost of low thrust chemical propulsion systems for orbital transfer of large space systems (LSS) from LEO to GEO or orbits that have equivalent ΔV requirements. The product of this effort was an analytic tool from which the benefits/cost of various engine systems can be determined. The effort was divided into three technical tasks with the following individual objectives.

TASK I - DEFINITION OF LSS MISSION CHARACTERISTICS

Task I determined the capture capability of each of three engine concepts for shuttle launched LSSs. Maximum payload launch capability of the shuttle was assumed to be 30,000 kg. The LSS is to be launched mated with the primary propulsion systems (PPS) in a single shuttle flight. Missions included only LSS's with deployed dimensions of over 30 m. NASA/commercial spacecraft that were too large for a single shuttle flight were only considered if they could be split into 2 or 3 launches with each section flown separately to GEO. A maximum of six launches were allowed for DOD spacecraft. A combination of mission acceleration range and payload mass with PPS capture envelopes gave preliminary mission capture results.

TASK II - BENEFIT VERSUS COST ANALYSIS MODEL DEVELOPMENT

Two distinct algorithms, benefit and cost, comprise the analysis model. The model calculates PPS costs and benefits values as a function of thrust. Major cost relationships for the PPS are based on subsystem masses. Costs such as those associated with production, launch, and deployment were incorporated in Life Cycle Cost (LCC). Ten benefit criteria follow a weighted criteria rating approach. Each PPS benefit is the sum of all criteria multiplied by their weighting factor. After the model was established it was exercised to predict areas that have the highest potential benefit gain from low thrust propulsion.

TASK III - SAMPLE PROBLEM SOLUTION USING BENEFIT VERSUS COST ANALYSIS TECHNIQUE

The model was fully documented including user instructions, a listing of the model, and detailed descriptions of the benefits and costing algorithms. Model input information on the two mission catalogs and three PPS was gathered. This data was used to exercise the benefit and cost analysis model and compared all three engine systems for the NASA/Commercial Catalog, DOD Catalog, and a combined NASA/Commercial and DOD Catalog. The results recommend an engine system and thrust range which minimize LCC and maximize benefit.

The objectives of Task I were to define a mission model, size a primary propulsion system (PPS) for each of three engine concepts, and combine these to produce a mission capture. Details of candidate engines were provided by NASA-LeRC and information required for the PPS was obtained from previous contracts. Large space system (LSS) mission details were obtained from many sources.

A. PROGRAM GROUND RULES

The following paragraphs present the groundrules for the engine/stage development and LSS mission model, respectively.

For the spacecraft sizing the Shuttle was assumed to deliver 30,000 kg (65,000 lbm) into low Earth orbit (LEO). Included in the 30,000 kg payload would be the LSS mated to the PPS and any necessary airborne support equipment (ASE). Specific design points for three LO_2/LH_2 engine concepts, an uprated RL-10 engine, an advanced engine, and dedicated low thrust engine, were supplied by NASA-LeRC and are listed in Table II-1. Performance data were also supplied and are plotted in Figure II-1.

Various LSS concepts and applications are currently being discussed but only those that are to be deployed in LEO and operated in GEO were considered for the NASA/COMMERCIAL list. The DOD missions also included spacecraft that had final orbits requiring transfer ΔVs similar to GEO requirements. In addition, only LSSs over 30 m diameter were included in the mission model, below this size conventional techniques for spacecraft construction and deployment can be applied. Originally only LSS/PPS payloads that could be launched in a single space transportation system (STS) payload bay were to be considered. Unfortunately, this would have resulted in only a few spacecraft in the mission model. To avoid too few missions spacecraft were included that could require a maximum of six Shuttles to launch a DOD spacecraft, while NASA and commercial missions were restricted to a maximum of three launches. If more than one Shuttle was required then payloads were divided equally, by mass, into the number of sections determined by STS capabilities and launched in the payload bay mated with its own PPS. Mating of the sections was assumed

to occur in GEO. The timeline for the LSS mission model catalog is from the current day to the year 2010. An original limit to the timeline of 1995 was relaxed to allow a more realistic scenario to be considered.

TABLE II-1 PPS ENGINE CHARACTERISTICS

	Uprated RL-10	Advanced Engine	Dedicated Low Thrust Engine
Thrust (Max) (Min)	66,720 N (15,000 lb_f) 6,672 N (1,500 lb_f)	66,720 (15,000) 4,448 (1,000)	4,448 (1,000) 890 (200)
O/F Mixture Ratio	6.0	6.0	6.0
I_{sp} (Max Thrust) (Min Thrust)	4,510 N sec/kg (460 lb_f sec/lb_m) 4,220 N sec/kg (430 lb_f sec/lb_m)	4,710 (480) 4,450 (454)	4,600 (469) 4,510 (460)
Area Ratio	205	640	400
Installed Length	1.40 m (55 in)	1.52 (60)	0.71 (28)
Mass	178 kg (392 lb_m)	177 (391)	40 (88)
Diameter, max	1.80 m (71 in)	1.63 (64)	0.53 (21)
NPSH, H_2O_2, kPa	13.2/26.6	3.05/6.66	3.05/6.66
NPSH, H_2/O_2, psid	1.9/3.9	0.44/0.97	0.44/0.97
NPSH, H_2/O_2, ft	64/8	15/2	15/2

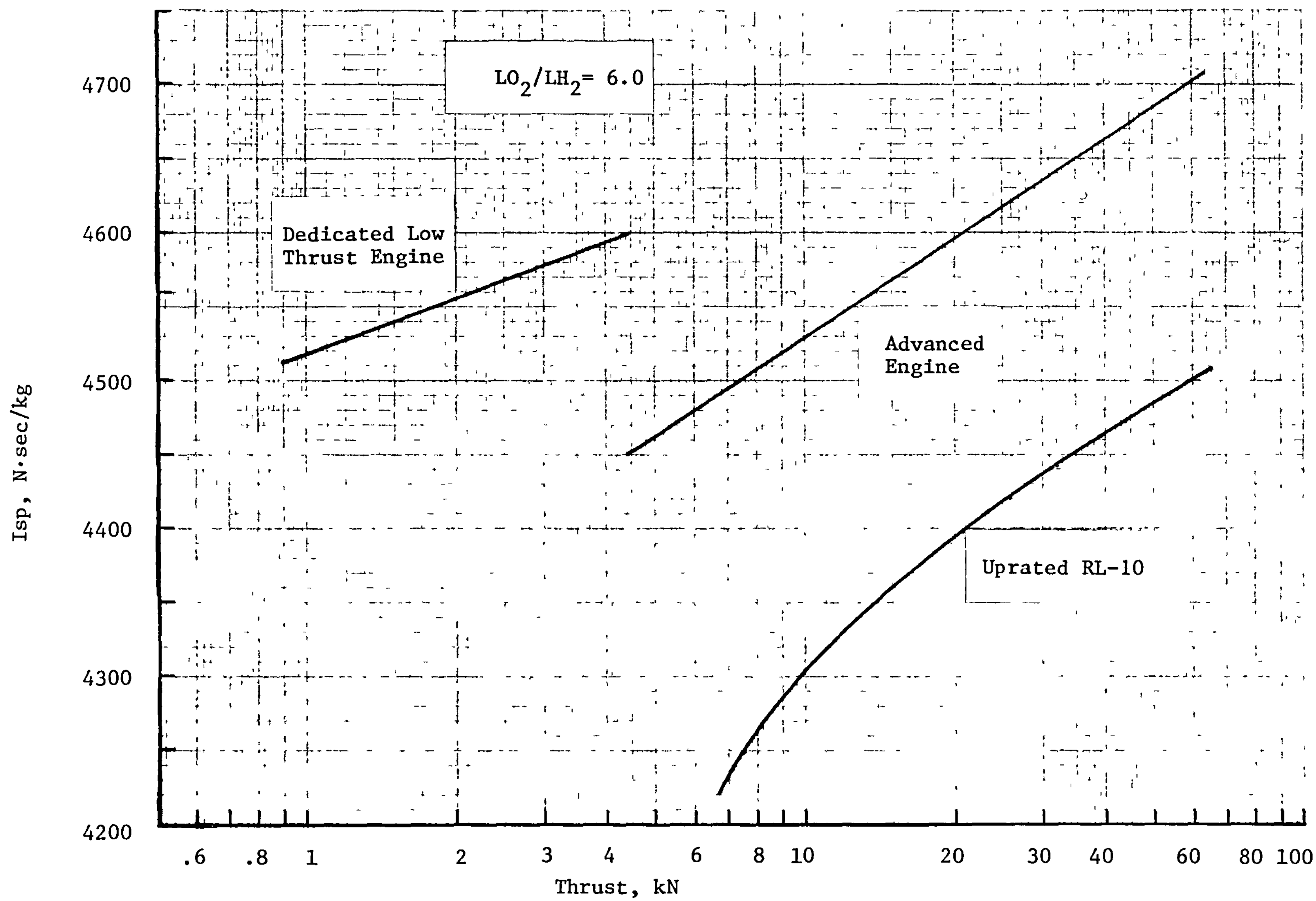


FIGURE II-1 PROJECTED PERFORMANCE FOR THE THREE STUDY ENGINES, SI UNITS

B. LSS REVIEW

1) Approach

The review of future LSS missions enabled the assembly of two mission models. Two mission models were defined, one for NASA/Commercial applications and one for DOD needs. This was necessary due to the classified nature of many military missions. Preliminary compilation of the projected NASA/Commercial LSS missions was accomplished with data from five main sources; Advanced Spacecraft Deployment System (ASDS) Study, completed by Martin Marietta Denver Aerospace (MMDA) for the AFRPL; a recently completed program by Boeing Aerospace for LeRC, Study of Electrical and Chemical Propulsion Systems for Auxiliary Propulsion of Large Space Systems (ECAPS-LSS); the MMDA IRAD Project D-54D, Large Space Structures; the Primary Propulsion/Large Space Systems Interaction (PP/LLSI) Study; and the DOD/STS Mission Integration Support Contract (formerly Payload Integration Contract). In addition a number of other references listed in Appendix A were used to obtain information. Military missions were found in the classified ASDS report and from the Military Space Systems Technology Model;Volume II, Systems Concept Options (MSSTM), prepared by the Aerospace Corporation (Reference 27, Appendix A).

Spacecraft included in the LSS mission catalog adhered to the study constraints described in the previous section. Preliminary choice of missions assumed that 8200 kg (18,100 lbm) was the upper limit for the payload mass deliverable to GEO. This estimate resulted from a calculation of maximum delivery capability of the advanced engine PPS. The limit on NASA/Commercial spacecraft of three launches per LSS excluded from this catalog any spacecraft whose total mass exceeded 24,600 kg (54,200 lbm). Similarly a single DOD spacecraft could not exceed 49,200 kg (108,400 lbm). In some cases a spacecraft has a total mass of less than 8200 kg and will require two Shuttle launches because of the low density of the packaged payload. This has been included when data was available. Each of the multiple launches of divided spacecraft was treated as an individual launch.

The inclusion of DOD missions required a separate listing of details due to their classified nature. Only a few details on the DOD missions have been identified in this report, the complete information for each spacecraft was reported to the AFRPL who are responsible for distribution. Each DOD mission is identified in enougn detail to be evaluated in Task III.

During the catalog development, the LSS fell into two areas of interest, 1) those spacecraft having applications similar to conventional satellite uses and 2) new applications possible only with spacecraft of large dimensions. Although many conventional applications are fairly predictable, the use of a LSS will provide a large step up over current capabilties. New applications are much harder to determine, thus any catalog of projected LSS missions must allow for new and innovative uses since it is very difficult to predict missions up to 30 years in advance. Past predictions for applications of new technologies were often underestimated so any prediction should attempt to allow for the unforseen. Therefore, LSS uses that would appear improbable by current standards are included in the mission model. If these seemingly lower priority missions do not materialize, the inclusion of these LSSs allow for yet unpredicted missions with similar spacecraft characteristics.

Since most of the GEO deployed LSSs are still conceptual, it was difficult to establish how firm each mission is. However, identification of applications which appear most viable was attempted. There are two major factors that will strongly influence the priority of these non-DOD missions. The first would be an economic concern, that is, an LSS is more likely to fly if the application is profitable - an example being commercial communication satellites that are now operating and providing an excellent return on investment. The second factor which would influence mission priority will be the research and development needs from the scientific community. However these needs are affected by government financing and are rather difficult to predict. Thus this catalog contains the flexibility to accommodate mission uncertainty. Military missions will generally be influenced by security needs first and funding second.

Information on unclassified missions was obtained from the open literature including studies conducted by the Aerospace Corporation, Boeing Aerospace, General Dynamics/Convair, and MMDA. Other references from various NASA centers and companies such as the Harris Corporation and Lockheed Missile and

Space Division supplied information on the antennas and structures. The classified documents, ASDS and MSSTM, supplied data for the classified missions.

The review of LSS mission requirements led to a revision in the original timeframe. Original guidelines called for development of a mission model for LSSs to be launched only during 1995 to 2010. With the concurrence of the NASA Project Manager, the lower limit of 1995 was dropped because it seemed highly probable that currently envisioned operational dates of many LSSs will slip. Therefore, missions that do not currently fall in the 1995 to 2010 timeframe could actually be launched within that period. Missions that have been proposed are very representative of spacecraft that may be required late in the timeframe. Perhaps the most important reason for a time frame revision was that all of the chosen spacecraft will require thrusts much lower than those available with currently projected Shuttle payload propulsion stages thus development of this PPS must precede the use of these groundruled missions.

2) Structures

Fourteen structural configurations were identified in the literature search (see Table II-2). The objective was to select from these concepts three configurations that represent the wide variety of structural and dynamic configurations. The majority of the fourteen concepts can be summarized into three generic classes of structure -- radial rib, hoop and column, and truss.

The wrap radial rib concept has the most efficient stowage density of all the radial rib configurations, is the most mature in design development, is capable of diameters to 200 meters, and is relatively light compared to other radial rib systems.

The wrap-rib antenna consists of a hollow, doughnut-shaped hub to which a series of radial ribs, formed to the shape of a parabola, are attached. A lightweight reflective mesh is stretched between these ribs to form the paraboloidal reflecting surface. The feed system is usually located at the prime focus of the paraboloid by one or more deployable support booms. A

sketch of the deployed wrap-rib antenna is shown in Figure II-2.

TABLE II-2 STRUCTURAL CONFIGURATIONS

CONCEPT	ORGANIZATION	DIAMETER* RANGE, m
UMBRELLA RADIAL RIB DOUBLE MESH ANTENNA	HARRIS (REF 4)+	3-25
WRAP RADIAL RIB ANTENNA	LOCKHEED (REF 18)	30-200
ERECTABLE RADIAL RIB ANTENNA	GENERAL (REF 13) DYNAMICS	30-200
RADIAL COLUMN RIB ANTENNA	HARRIS (REF 4)	20-100
ARTICULATED RADIAL RIB ANTENNA	HARRIS (REF 4)	20-40
MAYPOLE ANTENNA	LOCKHEED (REF 2)	30-300
HOOP & COLUMN	HARRIS (REF 4)	30-300
HOOP & COLUMN RADAR	GRUMMAN (REF 1)	30-200
EXPANDABLE TETRAHEDRAL TRUSS ANTENNA	GENERAL (REF 18) DYNAMICS	10-175
EXPANDABLE BOX TRUSS ANTENNA	MARTIN (REF 23) MARIETTA	10-250
SUNFLOWER SOLID PANEL ANTENNA	TRW (REF 16)	5-20
EXPANDABLE ASTROCELL MODULE	ASTRO RESEARCH/ LANGLEY	5-100
ELECTROSTATIC MEMBRANE	GRC (REF 22)	5-200
EXPANDABLE BOX TRUSS PLATFORM	MARTIN (REF 23) MARIETTA	5-100

NOTE: This table is from the PP/LSSI study, + "REF" number applies to Appendix A.

* Diameter limitations refer to single orbiter packaging with an orbit transfer vehicle.

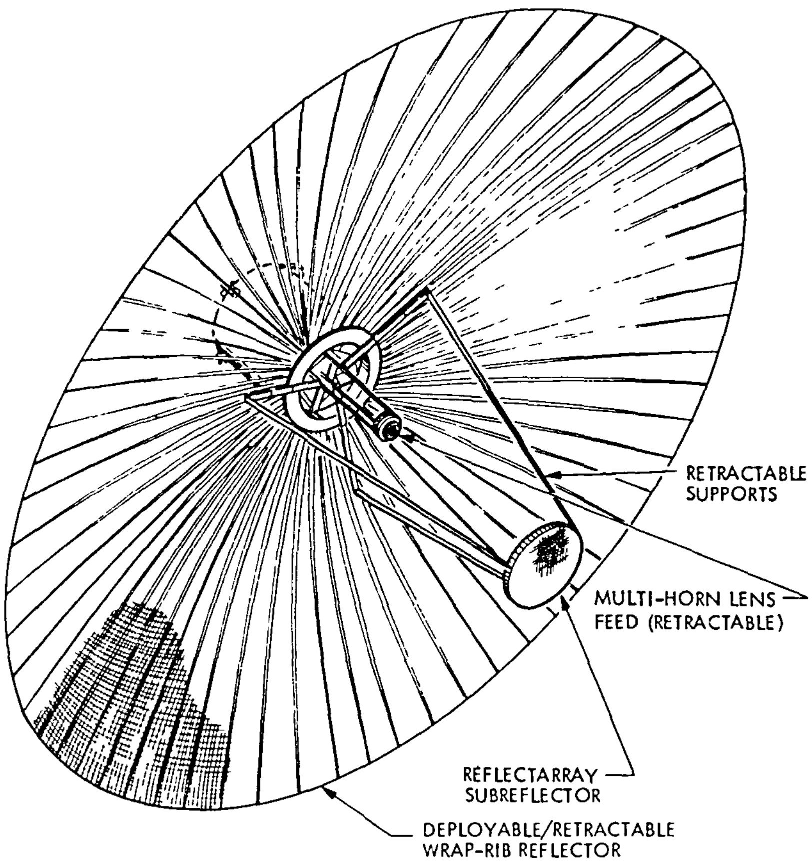


FIGURE II -2 TYPICAL LOCKHEED WRAP-RIB ANTENNA: DEPLOYED CONFIGURATION

The diameter range of the wrap radial rib is 30-200 meters where the actual maximum diameter is limited by the payload and stowage limits in the Orbiter. The primary mission application is a low frequency, large diameter reflector with a surface density of 0.05 kg/m^2, however larger surface masses are allowable.

For the hoop and column concept, the Grumman phased array and the Harris reflector concept were selected. The Grumman approach is typical of structure for arrays or solar collectors, and the Harris approach is typical of curved reflector surfaces (Figures II-3 and II-4).

The Grumman space-fed phased-array concept is intended for design up to 200 meters in diameter for operation at L-band or S-band. Grumman developed this concept to the point of a preliminary design for a 60 m diameter antenna and a 1.3 m diameter mechanical model. The mechanical model was used to demonstrate and evaluate the basic mechanical conceptual design.

The Harris Corporation hoop and column reflector antenna concept for self-erectable structures is intended for reflector designs up to 100 m in diameter (Figure II-4). This concept has been developed to the point of a preliminary design for sizes up to 45.7 m in diameter and a 1.8 m diameter conceptual demonstration model. This 1.8 m mechanical model was used to verify the basic conceptual design in addition to leading to solutions of the kinematic problems associated with deployment. The preliminary design has been complemented with the development of analytical techniques for prediction of antenna performance for larger size structures.

The fundamental elements of the support structure include the hoop; upper, lower, and center control stringers; and the telescoping mast. The reflector consists of the mesh, mesh shaping ties, secondary drawing surface, and the mesh tensioning stringers. The basic antenna configuration is a type of "may-pole", with a unique technique for contouring the RF reflective mesh.

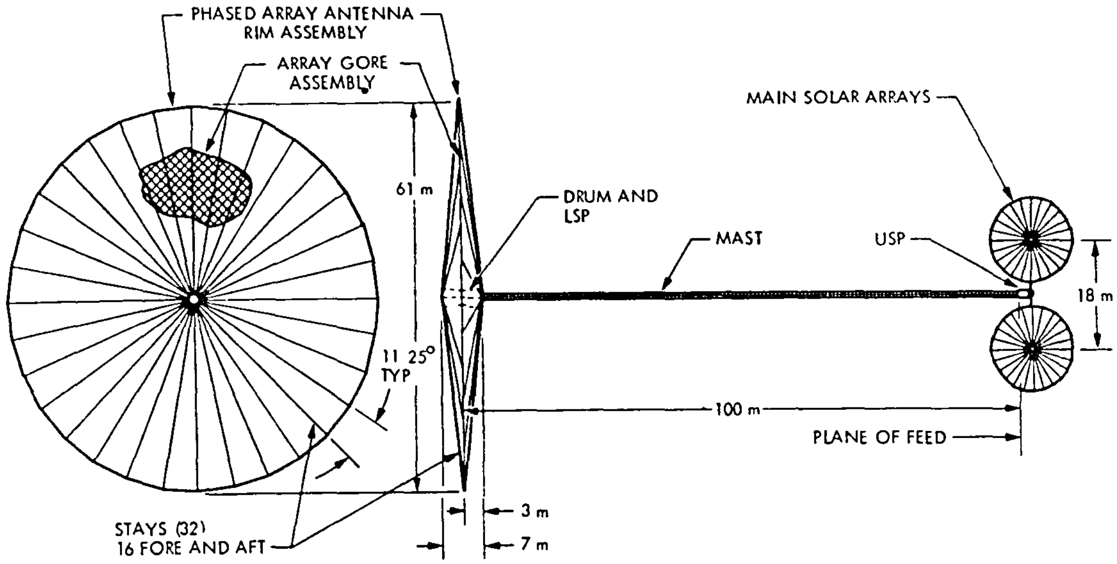


FIGURE II-3 BASIC STRUCTURAL ELEMENTS OF GRUMMAN PHASED-ARRAY CONCEPT

The diameter range of the hoop and column is 30-300 meters where the actual maximum diameter is limited by the payload and stowage volume in the Orbiter. The primary mission applications are a low frequency, large diameter reflector, a planar space based radar, and a planar solar array platform (surface mass density range of 0.05-0.15-0.40 kg/m^2).

Using data from the PP/LSSI study, for the truss concept, the box truss structure was selected, as shown in Figure II-5. This concept has the most efficient stowage density of all the truss concepts, is capable of diameters in excess of 200 meters, and is relatively light compared to other truss concepts.

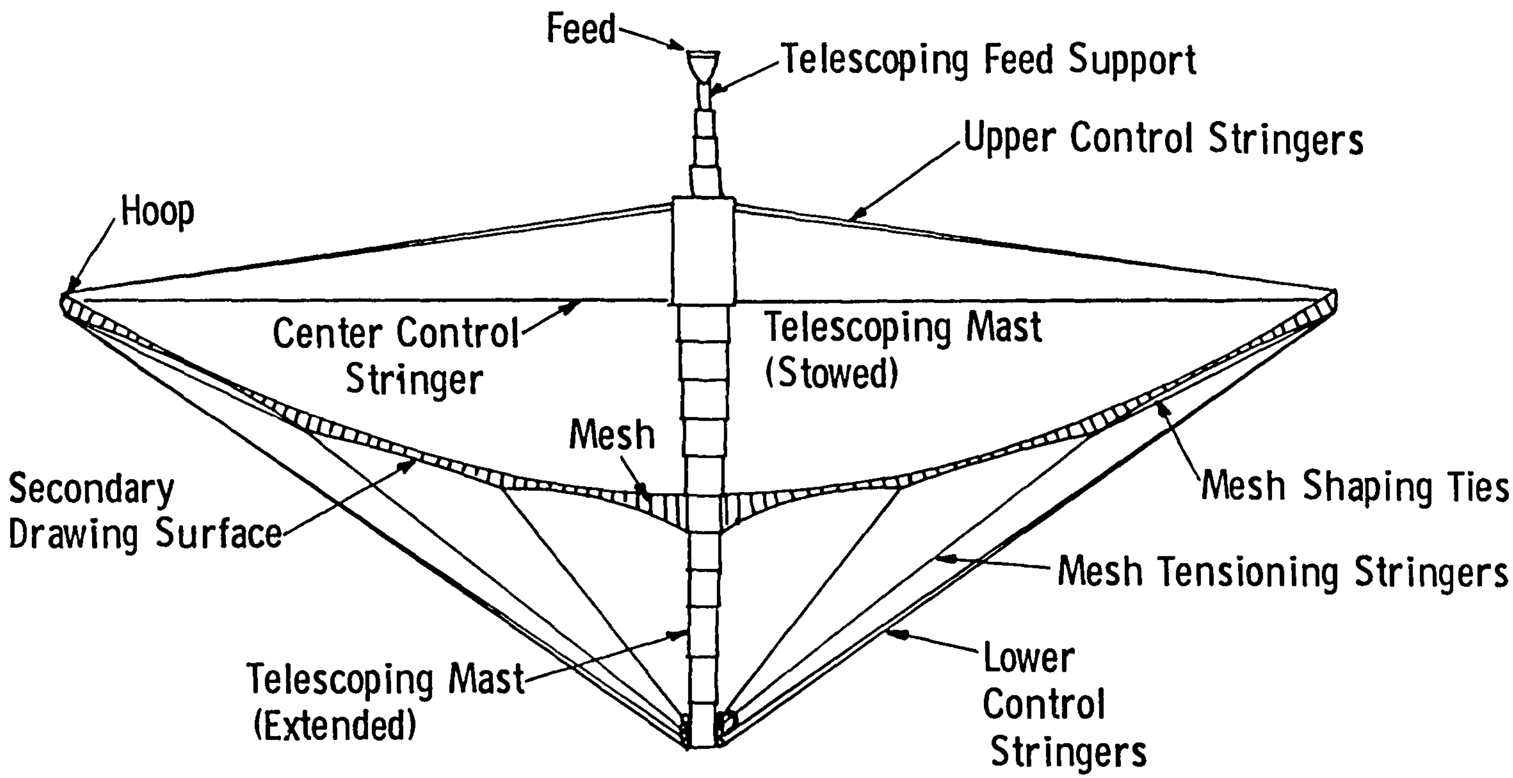


FIGURE II-4 HARRIS HOOP AND COLUMN CONCEPT

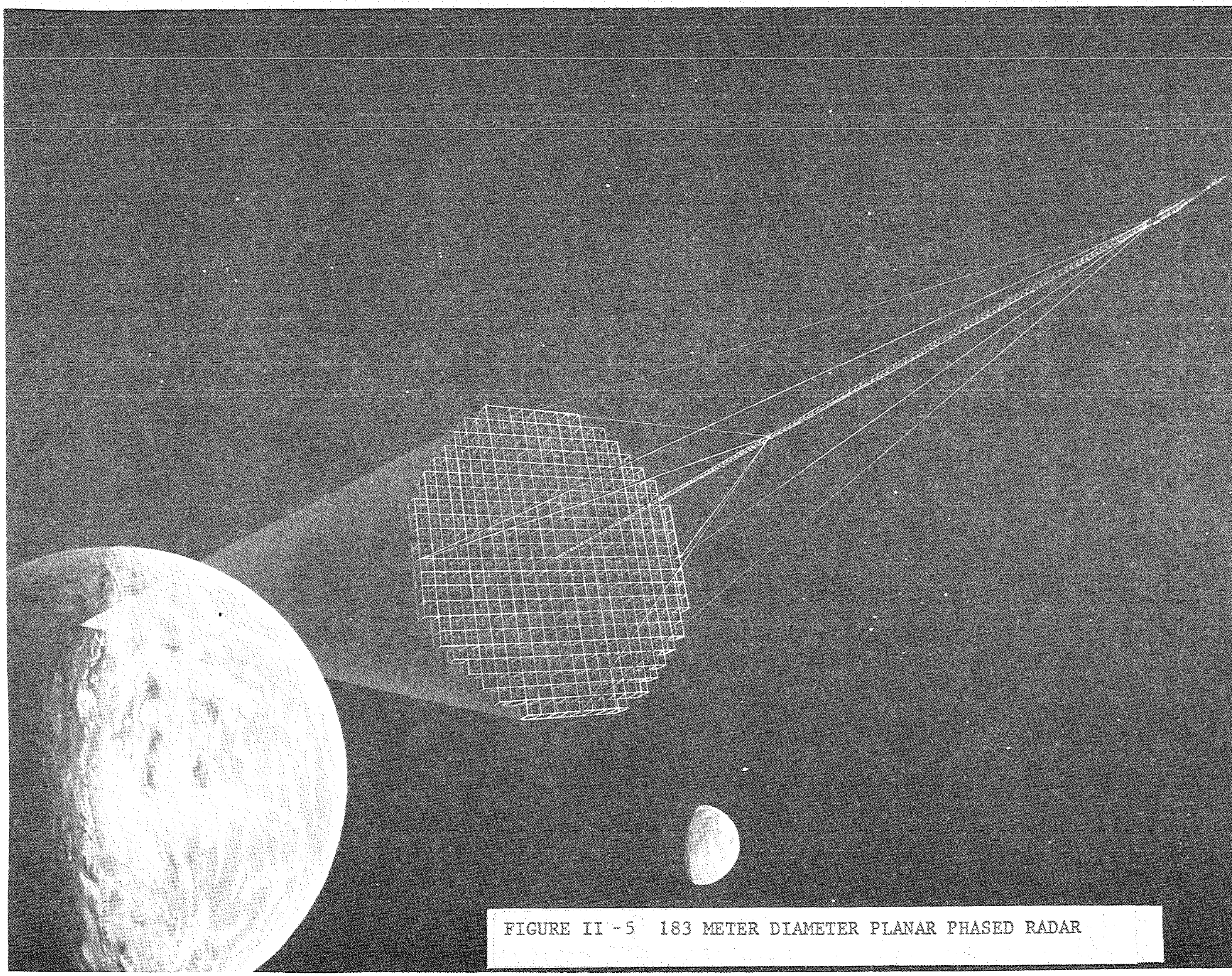


FIGURE II-5 183 METER DIAMETER PLANAR PHASED RADAR

Figure II-6 illustrates the basic concept's operating principle. Vertical members connect the front and back surfaces of the truss and carry support posts upon which the surface is mounted. Surface tubes, hinged in the middle, connect each vertical member to each adjacent member. Each truss square, composed of surface tubes and vertical members, is stabilized by diagonal tension tapes. For stowage, each surface tube folds about its mid-link hinge and the diagonal tapes telescope.

Structural deployment is accomplished in LEO near the Orbiter in a sequence of controlled steps. Following verification that each step has been completed successfully, the next set of rows is deployed. Symmetrical pairs are always deployed simultaneously to balance reaction forces. This preserves the deploying structure's attitude and center of gravity position.

The diameter range of the box truss is 30-200 meters where the actual maximum diameter is limited by the payload and stowage volume in the Orbiter. The primary mission applications are a low frequency, large diameter reflector, a planar space based radar, a planar solar array platform, and a science or communications platform (surface mass densities 0.05-0.15 - 0.40-3.42 kg/m^2).

Table II-3 presents a summary of the three LSS structure concepts which were selected as the baseline configurations for this study. Comparisons of the three classes are presented for single Shuttle diameter ranges, surface mass densities, point of thrust application, and applicable thrust to mass (T/m_f or acceleration) range.

3) Missions Identified

Review of the LSSs in the previous section revealed many applications for these large spacecraft, although not all were within the study guidelines. For example, some identified missions were for use in orbits other than GEO or the DOD high energy orbits; Earth-mapping radar, multi-national air-traffic-control radar, microwave-energy distribution, advanced resource/pollution observation, and some geo/atmospheric sensors. These

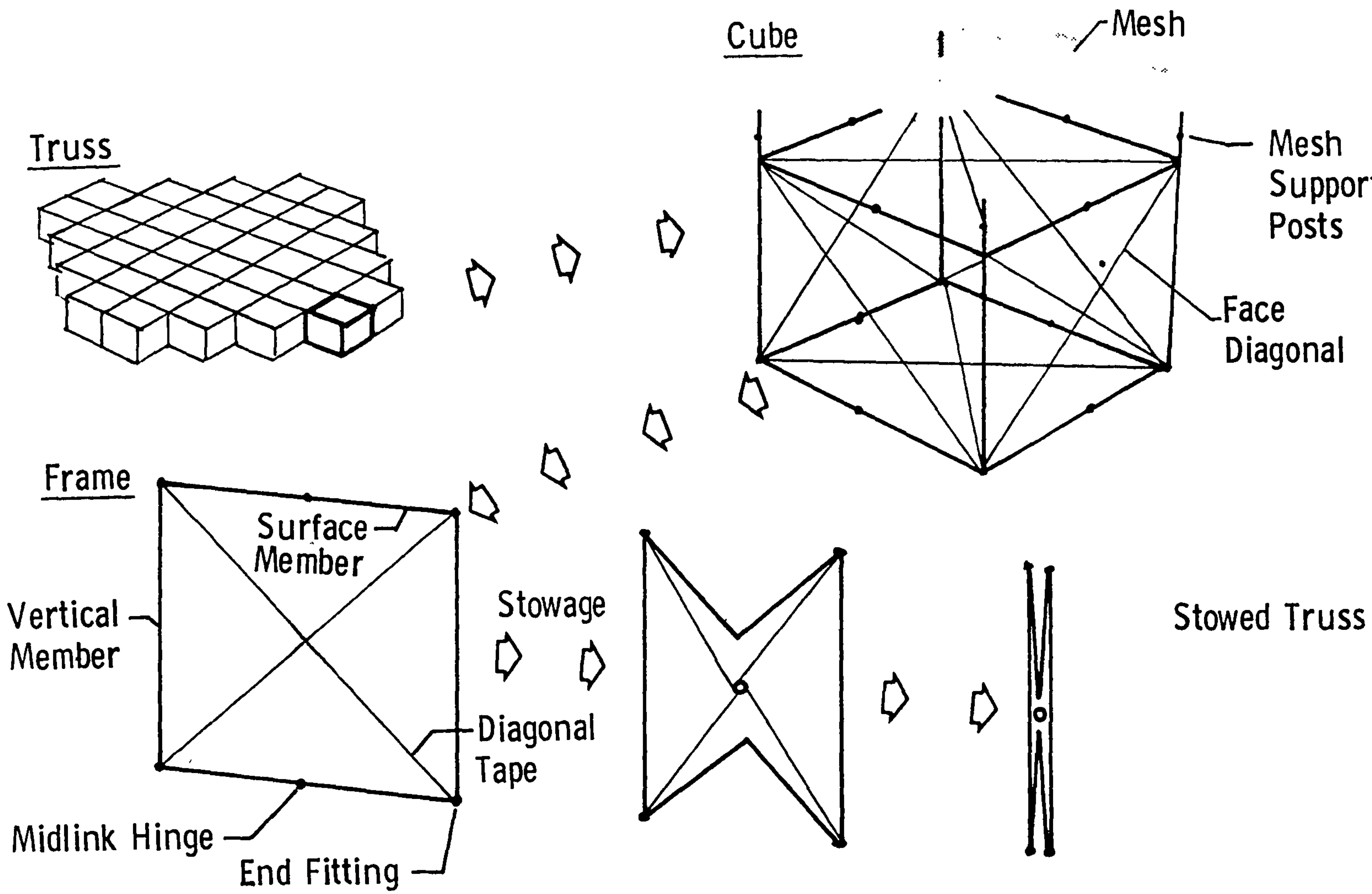


FIGURE II -6 DEPLOYMENT BOX TRUSS STRUCTURE

TABLE II-3 SUMMARY OF LSS CONCEPTS

Concept	Diameter Range (m)	*Surface Mass Density (kg/m^2)	Point of Thrust Application	T/m_f (g's)
Wrap Radial Rib	30-200	0.05-0.15	Hub	0.02-1.0
Hoop and Column	30-300	0.05-0.40	Aft end of mast	0.01-1.0
Expandable Box Truss	30-200	0.05-3.42	Center of Structure Normal to Plane	0.02-1.0

*Values are representative of typical missions:

0.05 for low frequency mesh type antennae
0.15 for radar antennae
0.40 for solar cell collectors
3.42 for high frequency antennae (aluminized honeycomb panels)

applications are not feasible in these orbits because of distance or orbital restrictions. Thus following the contract guidelines these uses were excluded. Once again, the classified nature of many military missions restricts the discussion in this report of those applications. However, many DOD missions were found to have similar applications to those in the non-military catalog. A brief review of the excluded missions would suggest that they would have little impact on the study results. Generally missions fall into certain orbit bands - low, medium, GEO and high. The high orbit requirements are DOD payloads and have ΔV needs similar to GEO deployment. Low and medium orbit requirements are small enough to be supplied by an integrated ACS/primary space storable system and thus would not affect this study.

Identified missions included a large percentage of communication type applications. This is to be expected since communication satellites require a stationary position in orbit, are profitable for commercial applications and critical for defense. The following list gives a brief description of each type of mission within the study guidelines, note more than one mission may be included in each application. Timeframes are the dates for initial operation of the satellites and are predicted from available literature.

Electronic Mail Transmission [Timeframe 1990-95]
- Speed up delivery
- Lower cost
- Service thinly populated areas
- 1 m ground station

Navigation Satellites [1992-2000]
- Provides relative position location to within 1 km (100 m for advanced concept)
- 1 required for CONUS
- Small inexpensive receiver

Geosynchronous Communication Platforms [1992-2005]
- Multiple antenna/frequency communication system
- Reduces costs/circuit
- Conserves orbital space and frequency use in GEO arc
- VHF through KU band with direct satellite to satellite links

Technology Development Platforms [1993]
- Provides long term test bed in GEO environment

Voting/Polling Wrist Set [2000]
- Allows for polling of voters opinion on major issues
- Could improve voter turnout
- Small wrist set receiver

Energy and Soil Monitor [1995]
- Monitors flow or consumption of energy by use of small one turn windings around transmission lines
- Measures soil conductivity to indicate soil moisture content

Marine Broadcast Radar (Coastal Anticollision) [2000]
- Single frequency radar transmission aimed at CONUS coastal areas for detection of marine hazards
- Ships will require conventional radar receivers

Orbiting Deep Space Relay Station [1995-2000]
- To supplement and or replace existing deep space network
- Will reduce dependence on foreign sites
- Can be used for VLBI

Personal Communications, Wrist Radio [1995-2000]
- Allows two-way voice telecommunications using small "Dick Tracy" wrist set
- Multichannel switching satellite that has many applications
- Could service up to 100,000 wrist phone wearers in each of 25 areas

Disaster/Police Communications Satellite [1995]
- Single antenna relay/switching station in the sky
- Could be combined with other functions on a single satellite

Burglar Alarm Relay Satellite [2000]
- Miniature sensors detect intruders and radiate a coded signal received by LSS in orbit
- 3 billion alarms per second could be processed over the whole United States

Space Based Radar [1995]
- Provides USAF with capability for long-range unjammable radar surveillance of aircraft, spacecraft, and missiles
- 5 satellites in service simultaneously

All of the missions identified in the above list are for deployable structures. But after space construction facilities have been built in orbit, most LSSs will be erectables or spacecraft completely constructed in space. Most deployables after this time will probably be replacement missions only.

C. MISSION CATALOG

1) NASA/Commercial Spacecraft

The literature search identified 16 missions for the NASA/Commercial catalog and these are described in Table II-4. All of the missions met the study guidelines, but it was necessary to separate this catalog into two subcategories. Missions 1 through 11 are those that are well within the delivery capability of all three propulsion systems. These are missions that can be accomplished with a single shuttle launch or by dividing into identical multiple launches with subsequent mating in GEO. The second category, missions 12 through 16, are those that must be transferred to GEO as a single payload and are close to the deliverable limits of one or more of the PPS. Regardless of mission, each orbiter will contain a PPS mated with either the complete spacecraft or the section to be flown to GEO. Spacecraft are described in Table II-4 by the following parameters.

Mission Number - The mission number is used for reference only in this study.

S/C Total Mass - This includes the mass of the structure, any hardware peculiar to that mission (example - switching mechanisms for communication satellites), solar cells, power distribution, and auxiliary control propulsion system.

Payload Dimension - The dimension that best describes that spacecraft. For a single antenna it is the diameter, for a planer array it would be the length of an arm, and for a platform it would be the maximum envelope dimensions of the spacecraft.

TABLE II-4 NASA/COMMERCIAL MISSION CATALOG

Mission (#)	S/C Total Mass, kg(lb_m)	Payload Dimension, m(ft)	Projected # of S/C Required	# of Shuttles Required per S/C and Mass of Each Section, kg	Acceleration Limits, gs	1st Year of Launch	Power Required, kW	Projected Minimum Lifetime, years	Remarks
Electronic (1) Mail Transmission -Demonstration	2400 (5300)	40 (130)	1	1/2400	1.0	1985	13	- -	-Does not require replacement -Single antenna -Low risk
Near-Term (2) Navigation Concept	1600 (3530)	48 x 0.5 (160 x 2)	1	1/1600	0.02-0.05	1987	1	5	-Will be replaced by mission 10 -Planner array
Demonstration(3) Geosyncronous Platform	4540 (10,000)	50 (165)	1	1/4540	0.2-1.0	1987	15	- -	-Detailed conceptual design done by GD/C -Modular antena design
Electronic (4) Mail Transmission	9100 (21,400)	2-61 (200) diameter antennas	2	2/4550 (1 Antenna per PPS)	0.02-0.1	1988	15	10	-Will require 2 replacements -2 61m antennas
Technology (5) Development Platform	3090 (6800)	1 x 1 x 50 (3x3x160)	1	1/3090	0.1-0.2	1988	160	10	-Long term test bed -Contains 30m diameter antenna -1 in service at a time
Full Capacity(6) GSO Communication Platform	8200 (18,100)	430 (1400)	6	2/4100	0.1-0.5	1990	30	Indefinite with Maintenance	-6 required for full global coverage -Modular or multiple antenna platform
Voting/Poling (7) Wrist Set	5900 (13,000)	46 (150)	1	1/5900	0.05-0.2	1990	90	5	-1 replacement -Likely to be combined with other functions eventually -Single antenna
Energy Monitor(8)	4540 (10,000)	46 (150)	1	1/4540	0.15-0.4	1990	23	10	-Single antenna
Orbital (9) Antenna Farm-America	6060 (13,400)	68 x 68 x 25 (220x160x80)	1	2/3030	0.05-0.2	1990	20	20	-8 antennas -Needs no replacement through 2010 -Multiple antenna farm

TABLE II-4 NASA/COMMERCIAL MISSION CATALOG (CONT'D)

Mission (#)	S/C Total Mass, kg(lb_m)	Payload Dimension m(ft)	Projected # of S/C Required	# of Shuttles Required per S/C and Mass of Each Section, kg	Acceleration Limits, gs	1st Year of Launch	Power Required, kW	Projected Minimum Lifetime, years	Remarks
Personal Navagation Wrist Set (10)	13,600 (30,000)	Cross 1700 x 5 per arm (5580 x 16)	2	2/6800	0.01-0.1	1993	2	10	-Phased array antenna -Assume it is possible to assemble both halves in GEO -Planner array
Marine Broadcast Radar (11)	6200 (14,800)	500 (1640)	4	2/3100	0.01-0.1	1995	25	10	-Broken into 2 sections due to large volume -Contains 2 150m antennas for direct communication
Orbiting Deep Space Relay Station (12)	7500 (16,000)	100 (330)	2	These spacecraft must be flown as a single payload, all are single antenna designs	0.05-0.2	1988	6	10	-2 required for VLBI -Replaces NASA present deep space network
Personal Communication (Wrist Radio) -Demonstration (13)	7260 (16,000)	50 (160)	1		0.25	1990	21	- -	-Switching functions tested
Disaster Communications Satellite (14)	8200 (18,000)	61 (200)	2		0.05-0.35	1990	15	5	-1 replacement -Likely to eventually be combined with other functions in single satellite
Police Communications Satellite (15)	8200 (18,000)	61 (200)	1		0.05-0.35	1990	- -	5	-1 replacement -Likely to be combined
Burglar Alarm Relay Satellite (16)	7260 (16,000)	61 (200)	1		0.05-0.35	1990	1	10	-1 replacement -Need not verified -Low risk

Projected Number of S/C Required - Usually most of these LSSs will require one spacecraft if only CONUS coverage is required. Number of spacecraft needed for other types of missions vary with application and/or the global area covered.

Number of Shuttles Required Per S/C and Mass of Each Section - Spacecraft are split into multiple launches if the mass exceeds the initial estimate of the maximum delivery capability of the PPS (missions 4, 7 and 10) or if information from the literature predicts that the volume required for the packaged payload would exceed that available in the orbiter bay (6, 9 and 11). The mass of each section in these cases is simply the total mass divided by the number of launches needed for one complete spacecraft.

Acceleration Limits - The thrust at the final engine burnout of a PPS orbital transfer is the most critical from a structures standpoint. As completion of the last burn occurs all of the usuable propellant has been expended, thus the acceleration is at a maximum. This value of the final acceleration (T/M_f) will actually determine the thrust level since the LSS will have a maximum acceleration beyond which structural damage will occur. For missions 1, 3, 6 12 and 13 acceleration values were found in the literature describing the mission. Acceleration values for other missions were estimated from the PP/LSSI Study. If acceleration limits were not available then a range was estimated from similar sized spacecraft presented in the PP/LSSI report. The lower value of the acceleration range represents the most conservative estimate or the maximum acceleration that the spacecraft can withstand while the upper limit is the least conservative estimate of an acceptable T/m_f. Thrust levels resulting in accelerations below the lower limit do not affect the structure but would impact the orbit transfer time and could significantly affect PPS performance and overall cost.

Design of these spacecraft will be highly dependent on the characteristics and restrictions of the launch and boost vehicles. It is expected that several of the spacecraft designs included in this catalog could require alterations on the basis of the final PPS design. Two areas possibly affected would be the spacecraft mass and the acceleration limits it is designed to withstand. The capability of the PPS to deliver a payload mass larger than that required for a particular mission would allow the designer to use more massive structures and thus increase overall strength. Although this strengthening would increase the total mass of the LSS it would also raise both the upper and lower limits of projected acceptable acceleration. This in turn would permit use of a higher thrust engine. But instead of assuming only state of the art capabilities and materials, the use of possible structural improvements could also increase the acceptable thrust range. Using this assumption, the increase in structural strength could allow a spacecraft to be designed to either 1) withstand higher accelerations - if the mass were to remain the same or 2) reduce structure weight - if the acceleration range needed to stay the same. An evaluation of how advanced structures could change the mission capture of each engine was evaluated and presented in Section IV-D-4.

First Year of Launch - Documents from which information was obtained for the mission model were written prior to STS-1 and with optimistic operational timeline for the Shuttle. The projected first launch dates for most of the misions were also optimistic. Since the initial operation of the Shuttle was delayed, a more realistic timeline needed to be projected. Therefore, it was decided to postpone all dates by five years, thus the earliest mission is now considered to be launched in 1990. This estimate considers delays in the initial launch of the STS, the reduction in the number of Orbiters to be purchased, increased turnaround time, and funding reductions. Revised timeline estimates may still be optimistic but, a five year postponement provides a more realistic projection.

Power Required - Obtained from literature and usually supplied by solar panels. Missions 5 and 7 have requirements considerable higher than those of the other fourteen missions. Of these two missions, number 5 is the technology development platform and will require a large amount of power for its experiments. Power for this mission will be supplied by deploying up to eight solar arrays of the type under development on the solar electric propulsion system program at the NASA Marshall Space Flight Center. Information on mission 7 is limited but with the 116 beams predicted, RF power output would be about 32 kW which would in turn require about 90 kW raw power input. Review of current and projected near-term technology provides an answer to the question of packaging these arrays in the orbiter. Folding arrays designed for solar electric propulsions (SEPS) are projected to have a power/surface-area ratio of 0.15-0.20 kw/m^2 before the end of the century. Additionally, a mass/power ratio of 15 kg/kw has been predicted for the SEPS array designed by NASA Marshall Space Flight Center. This would result in a mass of 2400 kg for the arrays on the technology development platform (Mission 5) and 1350 kg for the Voting/Poling satellite (Mission 7). Neither of these mass requirements would restrict either mission because both are well below the delivery capability of all three engines.

Volumetric packaging in the shuttle presents another concern. The NASA-MSFC 25 kw SEPS concept will package within two cannisters that are about 4 m long and up to 50 cm in diameter. Thus mission 5 would require 6 1/2 arrays of 25 kw type resulting in 13 of the packaged cannisters. This would not appear to represent a volumetric problem since these cannisters could be packaged with the platform in the Orbiter Payload Bay. Mission 7 would be even less restrictive since it would require 8 cannisters to be packaged with the Voting/Poling antenna.

Projected Minimum Lifetime is needed to predict replacement missions. These values are either supplied from the literature or estimated from similar spacecraft.

This collection of NASA and commercial missions was evaluated both independently and combined with the DOD missions for the mission analysis task. Comparison of LSS requirements and engine capabilities determined which missions each engine is capable of delivering.

2) DOD SPACECRAFT

As with the missions included in the NASA/commercial catalog, some information on the DOD spacecraft was not available and had to be estimated from data on similar concepts. In addition, only limited information on some missions can be reported due to the classification of the data. For these reasons some blanks appear in the DOD mission catalog.

A review of future DOD spacecraft requirements was performed among classified and unclassified documents and it is felt that the missions presented in this catalog are representative of future applications. The selection of military missions was conducted in the same way as the non-DOD catalog. Selected missions included not only concepts considered viable today but also those based on projected capabilities of LSSs through the year 2010. Future DOD missions have some uncertainties not associated with the NASA or commercial spacecraft. Military spacecraft are affected by both change in weaponry and strategic policy. For example, on policy, the amount that the DOD will switch to space observation or communications platforms could result in doubling the number of LSS in the Air Force inventory. Either of the two factors previously mentioned can greatly influence future plans, and for these reasons room must be left to allow flexibility in the mission catalog. This was accomplished by including all probable missions, allowing in the figure for missions to be dropped or added without adversely affecting the study conclusions. There is good reason to believe that any future missions not included in this catalog will be similar in structure and size to spacecraft that have been listed since generally a spacecraft is designed to be compatible with its launch vehicle and upper stage. This means that if a PPS were to be designed to the specifications of the LSS in the catalog used in this study, then missions planned in the future will, in turn, most probably be designed to meet the requirements of that PPS.

Projected lifetimes for military spacecraft are usually on the conservative side since an operational failure may result in serious security consequences. Since data on this spacecraft characteristic is usually classified information, one number was used for all missions, the seven year life projected for the space based radar system. Thus all DOD missions were projected to have a seven year lifetime, in addition it was assumed that each spacecraft would require a replacement. Missions selected for inclusion in the DOD portion of the catalog are shown in Table II-5 and have been identified as missions 17 through 29. Since only a few applications can be fully detailed due to the classified nature of much of the information, some spaces have been left blank. This table has a slightly different format from Table II-4 for this reason.

Missions 18 and 19 correspond to missions 16 and 14 in the NASA/commercial catalog, thus they can be fully detailed. These two military spacecraft have essentially the same applications as their civilian counterparts. Mission 17 was previously described in the Boeing Report (ref A-21) thus it was already fully defined. The rest of the missions in Table II-5 all have a limited amount of information available.

It should be noted again that these DOD missions are limited to 6 launches per single spacecraft and that not all missions are necessarily GEO operational.

Emphasis should again be placed on the fact that all of these missions, both NASA/Commercial and DOD, are very preliminary and some spacecraft currently have two or three designs for the same application. For those cases the most recent design was used. Although many of these concepts appear to be very advanced, from past experience one should be cautious in rejecting any "improbable" missions.

A graphic representation of the mission catalog is presented in Figure II-7. It shows the mass delivery capabilities required for each mission as well as the number of launches required. The numbers refer to mission number identified in Table II-4 and Table II-5. The number of STS

TABLE II-5 DOD MISSION CATALOG

Mission (#)	S/C Mass kg (lbm)	Payload Dimension, m (ft)	Projected Number of S/C Required	# of Shuttles Required Per S/C and Mass of Each Section, kg	Acceleration Limits, gs	Projected Minimum Lifetime, years
Space Based Radar- Far Term (17)	7000 (15,000)	100 (330)	4	1/7000	0.05-0.1	7+
Security (18) Surveillance of Unmanned Sites	7260 (16,000)	61 (200)	1	1/7260	0.05-0.35	7
Distress Signal (19) Pinpointing	8200 (18,000	62 (200)	2	1/8200	0.05-0.35	7
(20)	14,660 (32,300)	*50(150)	4	2/7330	0.05-0.2	7
(21)	36,650 (80,800)	*	5	5/7330	0.05-0.2	7
(22)	5,900 (13,000)	*	2	1/5900	0.05-0.2	7
(23)	45,400 (100,000)	*	2	6/7570	0.1-0.2	7
(24)	4,540 (10,000)	*	4	1/4540	0.05-0.2	7
(25)	11,340 (25,000)	*·	5	2/5670	0.05-0.2	7
(26)	45,400 (100,000)	*	3	6/750	0.05-0.2	7
(27)	7,000 (15,000)	*	3	1/7000	0.05-0.1	7
(28)	45,400 (100,000)	*	2	6/7570	0.05-0.2	7

\+ Assumed value, also each mission will require a replacement

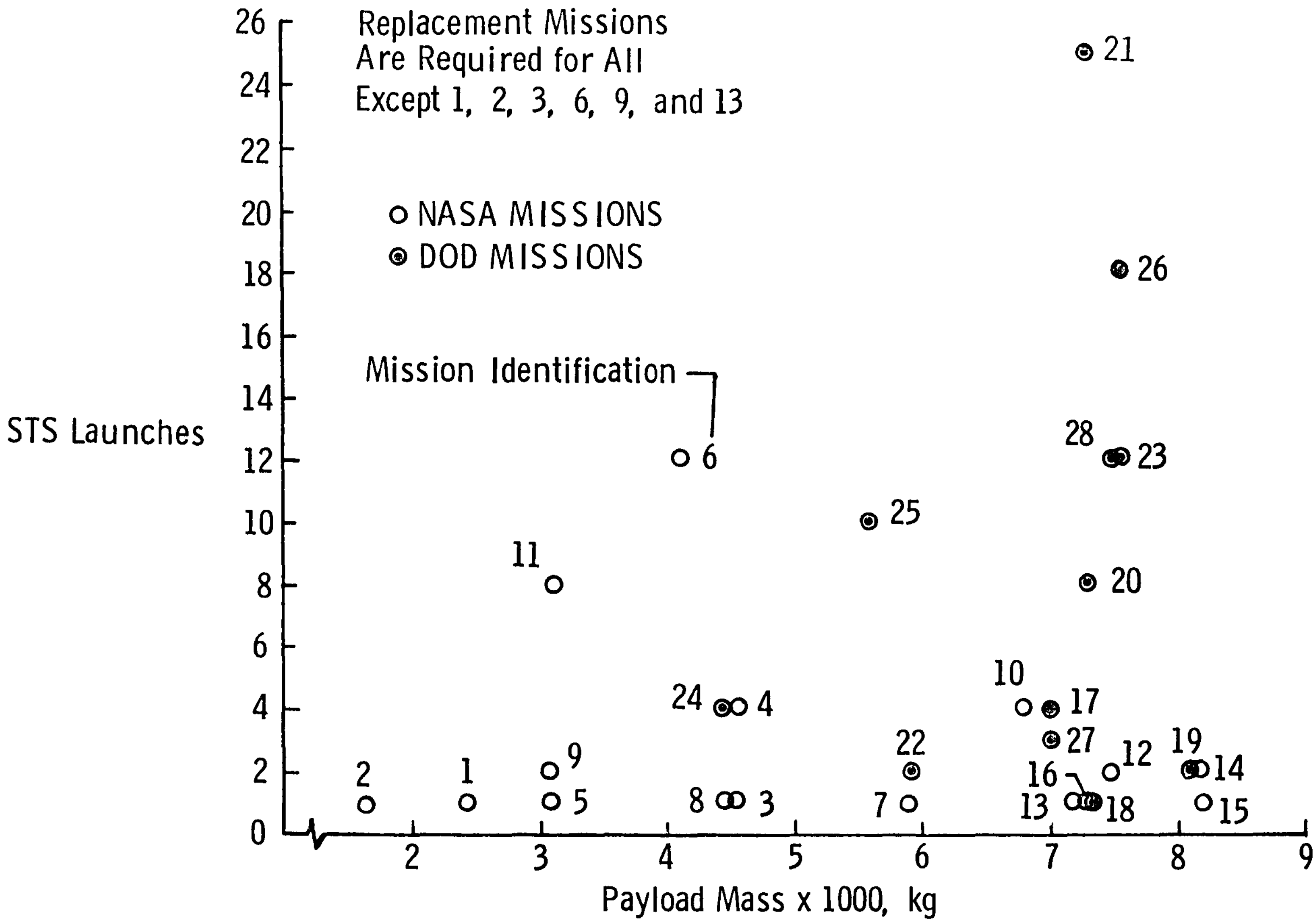


FIGURE II-7 MISSION CATALOG REQUIREMENTS

launches was obtained by multiplying the projected number of S/C required by the number of Shuttles required per S/C. Replacement spacecraft were assumed to be required for operational satellites whose estimated lifetime would indicate a failure before the year 2010. Lifetimes for many spacecraft assumed servicing in GEO, if this is not possible then the number of replacement missions would more than double.

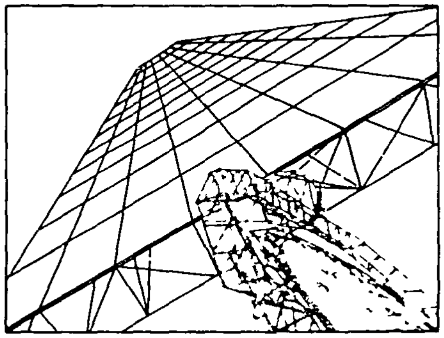
No pattern seems to emerge from Figure II-7 since the missions are spread over a range of masses and no single requirement dominates the graph. The figure does indicate that all of these missions are within the calculated payload mass capabilities of the engines under investigation. However, the graph does not address the effects of payload acceleration limits. These effects could only be evaluated after the PPS sizing was completed.

D. SPACECRAFT CLASSIFICATION

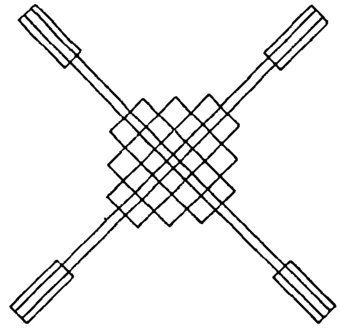
This portion of the task was not as important as originally considered because of the mission capture approach developed. That is, instead of dealing only with a class of structures, each mission was considered individually. In the recent ECAPS-LSS study completed for NASA-LeRC, an approach was used to classify LSS by shape. The major categories were single antennas, planar arrays, and antenna platform concepts (see Figure II-8) In addition to these major generic classes they were also broken down into sub-classes. For our study only the planar array and antenna platforms were subdivided since the single antenna class only contained deployable antennas.

Major breakdown of LSS was by application however, since it is the most important way to categorize the missions. For the applications identified in this study five major classes were chosen as shown in Table II-6. The largest portion of the mission model falls under the heading of communications, this was followed by navigation/maritime radar, space based radar, exploration/scientific, and Earth observation. Table II-6 applies to many DOD missions as well as the NASA/Commercial catalog and the list gives the ranges of characteristics for each application.

Planer Array

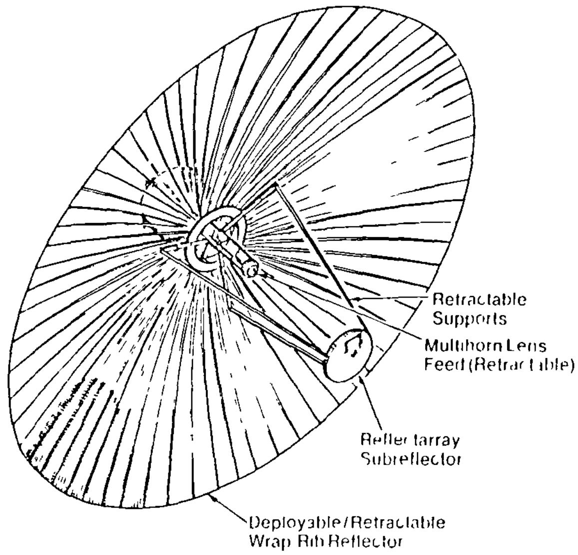


Flat Plate

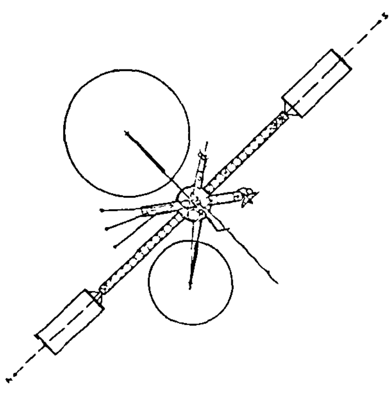


Cross Structure

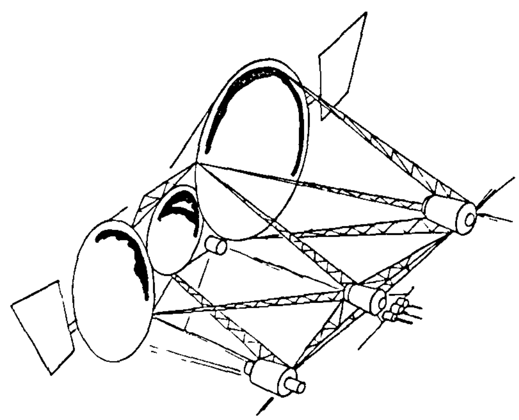
Single Antenna



Antenna Platforms



Modular Antenna Farm



Multiple Antenna Farm

FIGURE II -8 GENERIC CLASS SUBDIVISION

TABLE II-6 SPACECRAFT CLASSIFICATION

CLASS	DIMENSIONS	OPERATING FREQUENCY GHz	STRUCTURE TYPES	SURFACE DENSITY KG/M^2	POWER REQUIRED kW
Communication Single Antennas	45-60m Diameter	1-5	Wrap Radial Rib; Hoop/Column	0.05	5-90
Antenna Platforms	30-170m Wide 50-450m Long; 3-100m Diameter	1-20	Truss; Wrap Radial Rib	0.05 for Low Frequency 0.3 for High Frequency	15-30 (Solar)
Navigation/Maritime Radar Planer Array, Cross Structure	48-1700m Long Arms; 50-150m Diameter	10-12	Truss; Wrap Radial Rib	Phased Array 0.15 Antennas 0.05	1.25 (Solar)
Space Based Radar Single Antenna	180m; 270m Long Mast	1.5	Box Truss; Wrap Radial Rib, Hoop/ Column	0.15	50 (Nuclear Power)
Exploration/Scientific Single Antenna	100m	3	Wrap Radial Rib; Truss	0.05	6 (Solar)
Modular Antenna Platform	50m Long 30m Diameter	1-17	Truss; Rib; Hoop/Column	0.05 for Antenna 0.40 for Solar Panels	160 (Solar)
Earth Observation Single Antenna	40-60m Diameter	1.5	Rib; Hoop/Column	0.05	23 (Solar)
Modular Antenna Plat- form	50m Long 30m Diameter	1-17	Truss; Rib; Hoop/Column	0.05 for Antenna 0.40 for Solar Panels	160 (Solar)

E. PROPULSION SYSTEM SIZING

The size of the PPS was determined by engine performance characteristics and the maximum possible mass of the LSS delivered to GEO. Engine details supplied by NASA-LeRC have already been shown in Table II-1 and Figure II-1. Vehicles were sized generically at the maximum combined PPS/LSS mass of 28,000 kg and at 20,000 and 12,000 kg. These total values of system mass provided a performance envelope of final acceleration and deliverable payload mass for each engine. The upper limit of 28,000 kg excludes the 1545 kg for the ASE and 455 kg for two manned maneuvering units. The mass of the ASE is slightly lower than the figure used in previous studies (LTPS, PP/LSSI, ASDS) but more detailed analyses suggest the new value is correct.

Eight perigee burns and one apogee burn were used for all stage sizing. This strategy was used across the entire thrust range of each engine even though high thrust stages (greater than 22,250 N) do not benefit significantly from multiple perigee burns. Since emphasis of most LEO deployed LSS missions was for low thrust (final accelerations of less than 0.1 g) this assumption did not bias the results significantly towards lower thrusts.

Engine characteristics along with information from the ASDS and LTPS studies defined the PPS. Conceptual stage designs were sized over each engine thrust range. The basic vehicle is shown in Figure II-9 with a list of the hardware masses, exclusive of tankage equipment. Most stage characteristics were those defined in the LTPS study. The stage diameter of 4.42m (14.5 ft) allowed for a maximum tank diameter of 4.27m (14 ft) and an ellipsoidal/toroidal tank configuration minimizes stage length. For a LO_2/LH_2 vehicle, the tank arrangement shown in the figure is about 2.5m shorter than a similar capacity stage using a conventional ellipsoidal/ellipsoidal configuration. Values in Table II-7 were obtained from the ASDS and LTPS studies and included in the PPS mass along with all propellants required, and tankage systems.

Propellant requirements were calculated using the computer program, PROP (a summary flow chart of this program is shown in Figure II-10). This program also determines the tankage and insulation mass needed for the calculated amount of propellants. Boiloff and usable propellants are computed by the

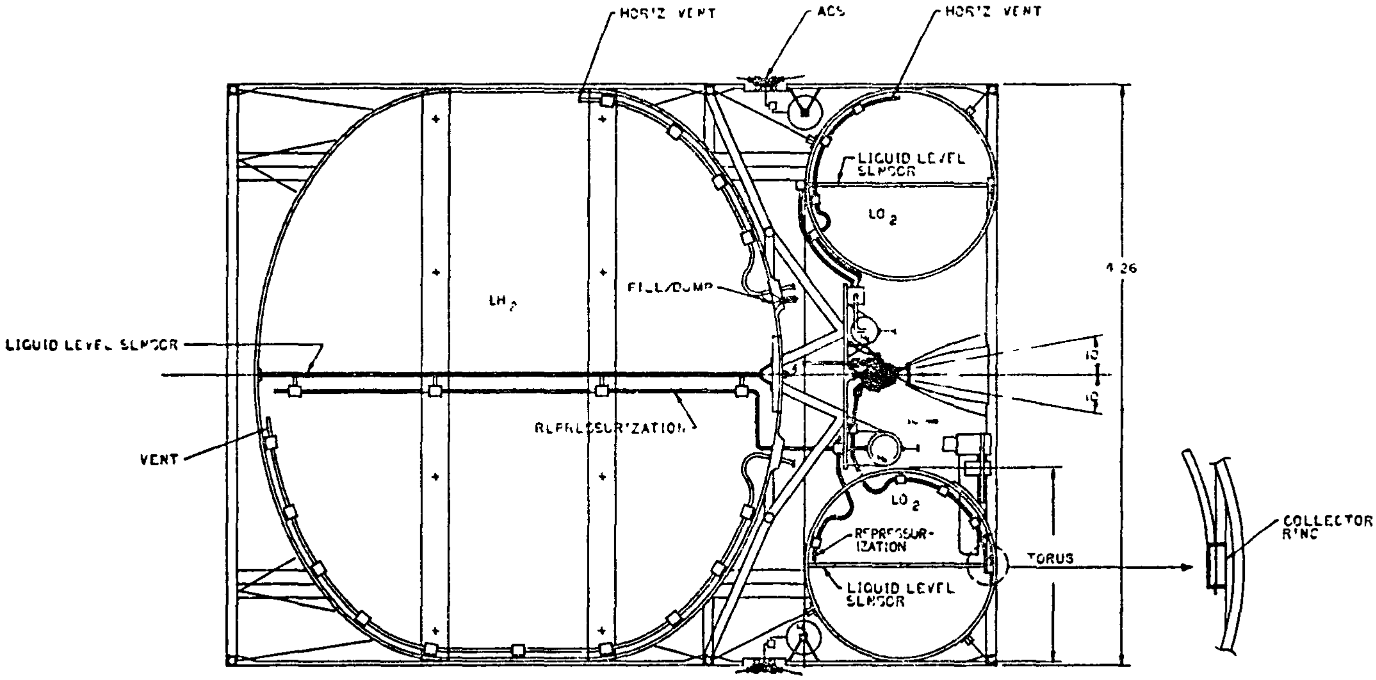


FIGURE II-9 PPS PRELIMINARY DESIGN

TABLE II-7 NON-TANKAGE HARDWARE MASSES FOR THE PPS

Components		Mass, kg	(lbm)
(Avionics, data management, computer fuel cell, and communications)		340	155
Structures (external shell, Shuttle I/F equipment, equipment mounting, etc.)		460	209
Propellant Feed System		170	77
ACS Components and Propellant		320	145
Purge System and Thermal System (not including insulation)		120	55
Engine Mounts and Supports		45	20
Components and Lines		25	11
	Total	1480	672

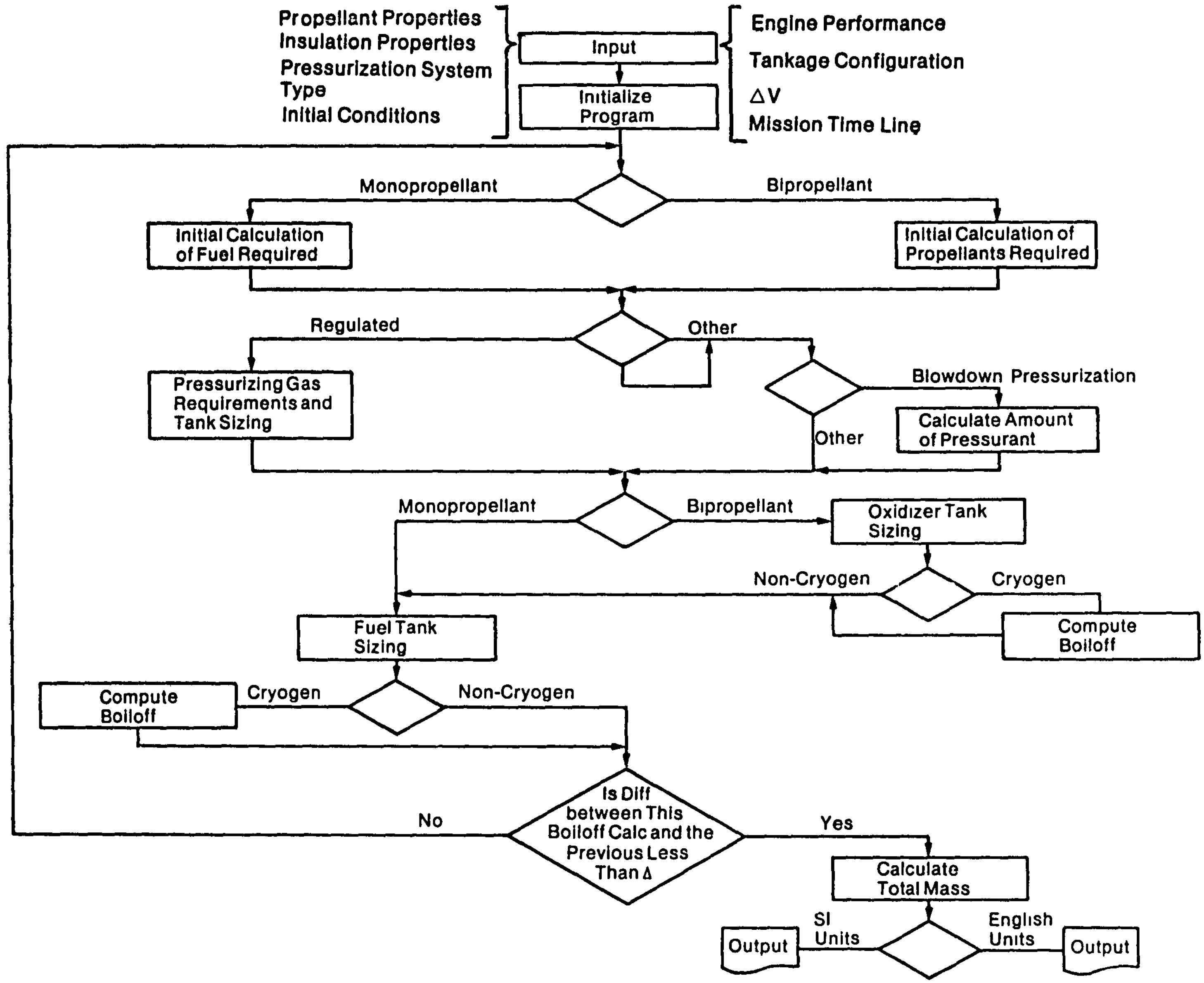


FIGURE II -10 PROP PROGRAM SUMMARY FLOW CHART

program and about 3% was added for trapped propellants and inaccuracies in filling and draining. Optimum multilayer insulation (MLI) thicknesses were calculated to be 5.1 cm (2.0 in) for the LO_2 and 4.5 cm (1.8 in) for the LH_2 tanks. Pressurization system masses, for each engine NPSH, were taken from an ongoing NASA-LeRC study entitled Propellant Expulsion and Thermal Conditioning Study [NAS3-22650] and are shown in the Table II-8.

Table II-8 Pressurization System Mass

	Mass Penalty for Pressurization System, kg (lbm)	
Engine Type	LO_2 - Helium Bubbler	LH_2 - Thermal Subcooler
RL-10	145 (320)	100 (220)
Advanced or Dedicated Low Thrust Engine	127 (280)	82 (180)

The basic configuration of all three PPS was identical, the only difference being the size and delivery capability of each stage due to the engine. Sizes predicted by PROP reflect these variations in performance. Outputs from the computer routine included the maximum acceleration at the end of the circularization burn which is the T/m_f and the mass of the vehicle at the time of STS liftoff. Using this data, Figure II-11 was plotted to show the GEO delivery capability of each PPS. Payload mass was found by subtracting the predicted vehicle mass from the total initial mated PPS/LSS mass. It was assumed the lower initial mated masses (those less than 28,000 kg) would be achieved by offloading propellants from the full size vehicle. Figure II-11 shows that the final thrust/mass (T/m_f) ratio increases as the total PPS/LSS mass decreases. This is a first order effect of increased Isp at higher thrusts. Final acceleration levels for the three engines cover the acceleration ranges identified in the mission catalog. In some cases a payload can be delivered by a vehicle that has a lower T/m_f than the most conservative value (lowest) detailed in the catalog. Thus an 890 N thrust

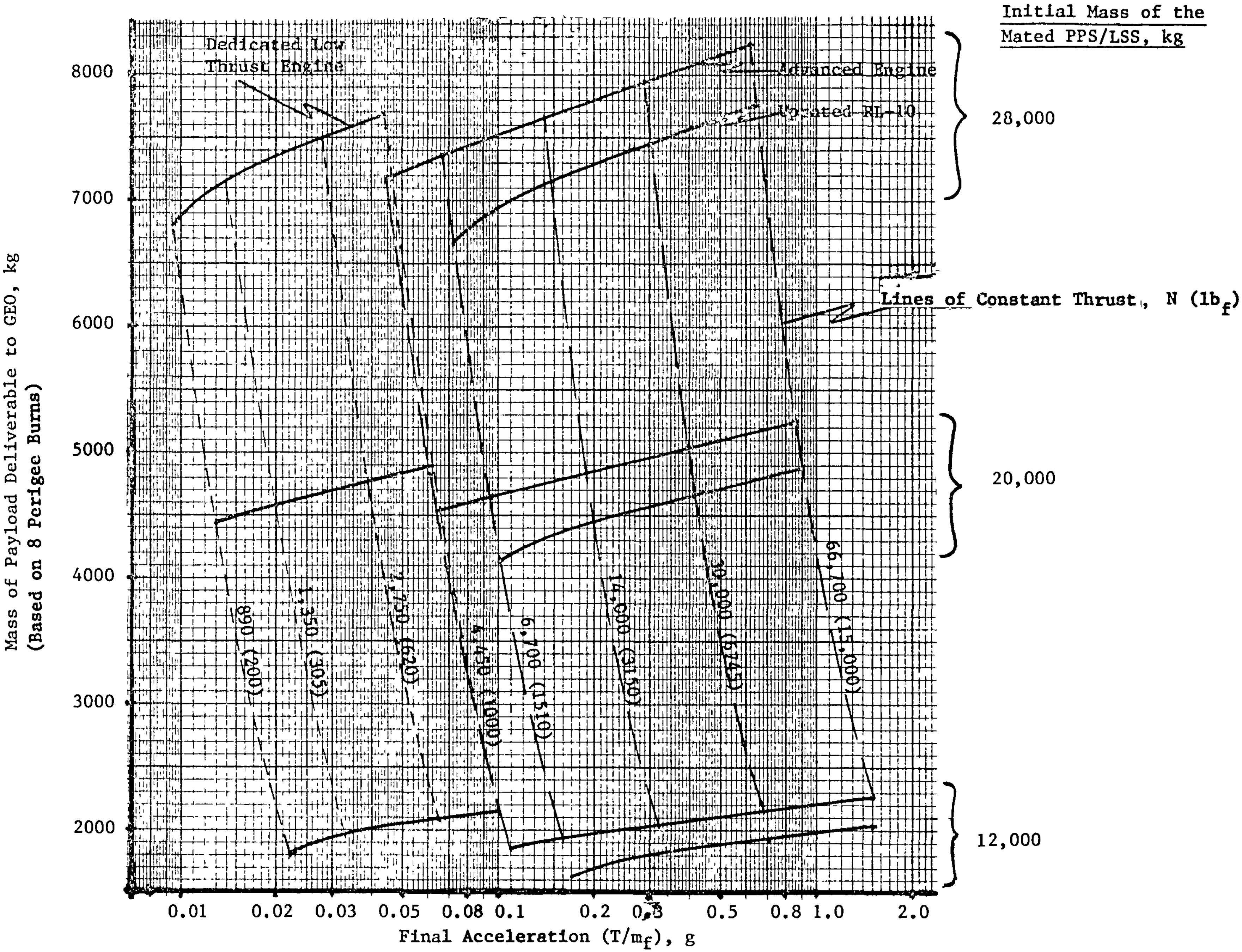


FIGURE II -11 PPS PERFORMANCE FOR VARIOUS MASSES OF THE MATED PPS/LSS

level could possibly be used to deliver a spacecraft that has a lower acceleration limit of 0.1 g, if the mass of the LSS does not exceed the maximum deliverable mass at that thrust level.

An initial estimate of each engine mission capture is plotted in Figure II-12. This graph was produced by combining Figure II-11 with the bracketed acceleration ranges for each mission in Table II-4 and Table II-5. Each black horizontal bar represents the range of final accelerations within which the actual spacecraft will reside. The left end of the bar is the most conservative estimate of the acceleration the LSS structure will be able to withstand and thus the right end point would be the least conservative. A range is necessary since none of these missions have been fully analyzed as of yet. If it is assumed the most conservative estimate is correct then mission 9 will require a thrust level available only with the dedicated low thrust engine. On the other extreme, if the least conservative estimate is the correct value for this mission, then it could be delivered by a PPS using either the advanced engine or the uprated RL-10. Missions 14, 15, and 19 fall outside of all three engine performance envelopes (these exceed payload delivery capability) and mission 2 only falls within the dedicated low thrust engines envelope. Although missions such as number 3 have higher acceleration ranges than the dedicated low thrust engine reaches, this engine can still capture this mission since there is no significant differences delivering the required payload mass to GEO at the lower thrust level. Figure II-12 shows that missions 11 and 1 through 9 are well within the delivery capabilities of any of the three engines, thus only acceleration limits need to be considered for these missions. Higher thrusts generally produce higher final accelerations but also provide improved engine performance and allow for more efficient orbit transfers, for this reason it is preferred to be able to use the highest thrust allowable for the PPS. However from the structural point of view lower accelerations, or lower thrusts, are preferable. If the most conservative final acceleration is taken then the mission capture for 11 and 1 through 9 is always improved by lower thrusts, since the lowest acceleration value for these mission is not a limiting factor.

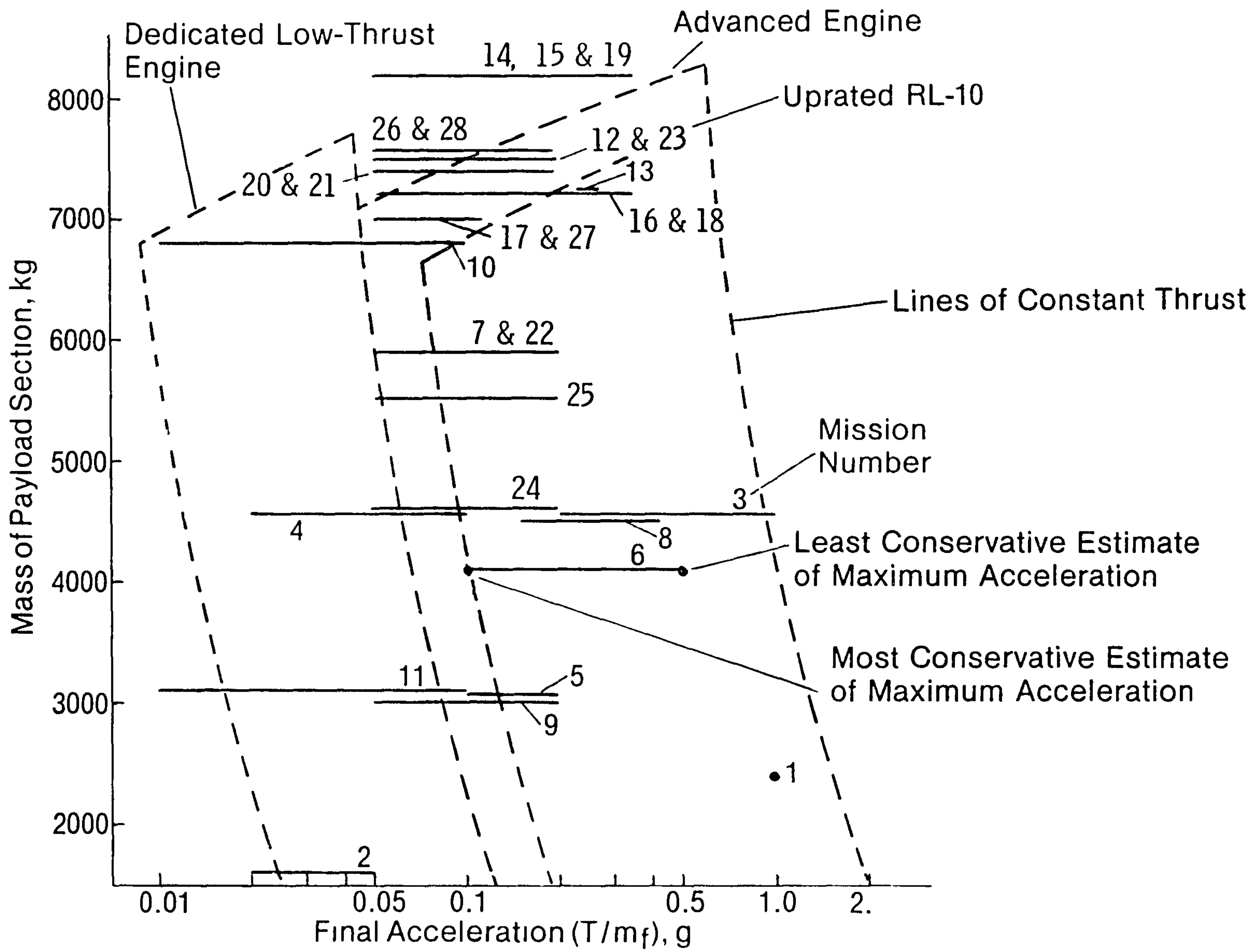


FIGURE II-12 MISSION CAPTURE ENVELOPES

The main effect of relaxing the lower acceleration level is to increase orbital transfer time. An example can be shown from Figure II-12 and Figure II-13. Mission 6 has a lower limit of 0.1 g but the payload mass can still be delivered at a T/m_f of 0.01 g. This decrease in acceleration drives the trip time from 30 hours to about 50 hours (see Figure II-13). This would result in a small increase in boiloff and an increased attention span for ground control with no significant effect on the LSS. In contrast, Mission 12 can only be delivered by the advanced engine or the dedicated low thrust engine above a certain thrust level. Below that minimum level, that mass cannot be delivered to GEO because of engine performance.

The possibility of mission capture exists for an engine at final acceleration levels less than the LSS "design point" acceleration. However, penalties associated with capture at lower final accelerations are increased transfer time, increased boiloff and degradation of electronics.

A worst case to illustrate these lower acceleration penalties is a heavy payload with a high final acceleration which requires essentially no propellant off-loading. This worst case creates more boiloff than a light payload which requires less propellant. A mission which represents the worst case is Mission 13. The personal communication mission has a payload of 7260 kg and a final acceleration of 0.25 g. The transfer time from LEO to GEO for this spacecraft at 0.25 g is 28 hours. Boiloff for this combination of payload and final acceleration is approximately 850 kg (1870 lbm). The acceleration requirement for this mission will be relaxed as far as payload capability permits for each engine. Specific effects of relaxed acceleration required for the uprated RL-10, advanced engine, and dedicated low thrust engine are summarized below.

The lowest possible final acceleration for the uprated RL-10 is 0.2 g. This lower acceleration increases transfer time by 0.5 hours and boiloff by 4 kg. The advanced engine can deliver the spacecraft at an acceleration equal to .05 g. This final acceleration corresponds to a transfer time increase of 3 hours and to a boiloff increase of 20 kg. The engine that has the greatest penalty is the dedicate low thrust engine. While the transfer time increases

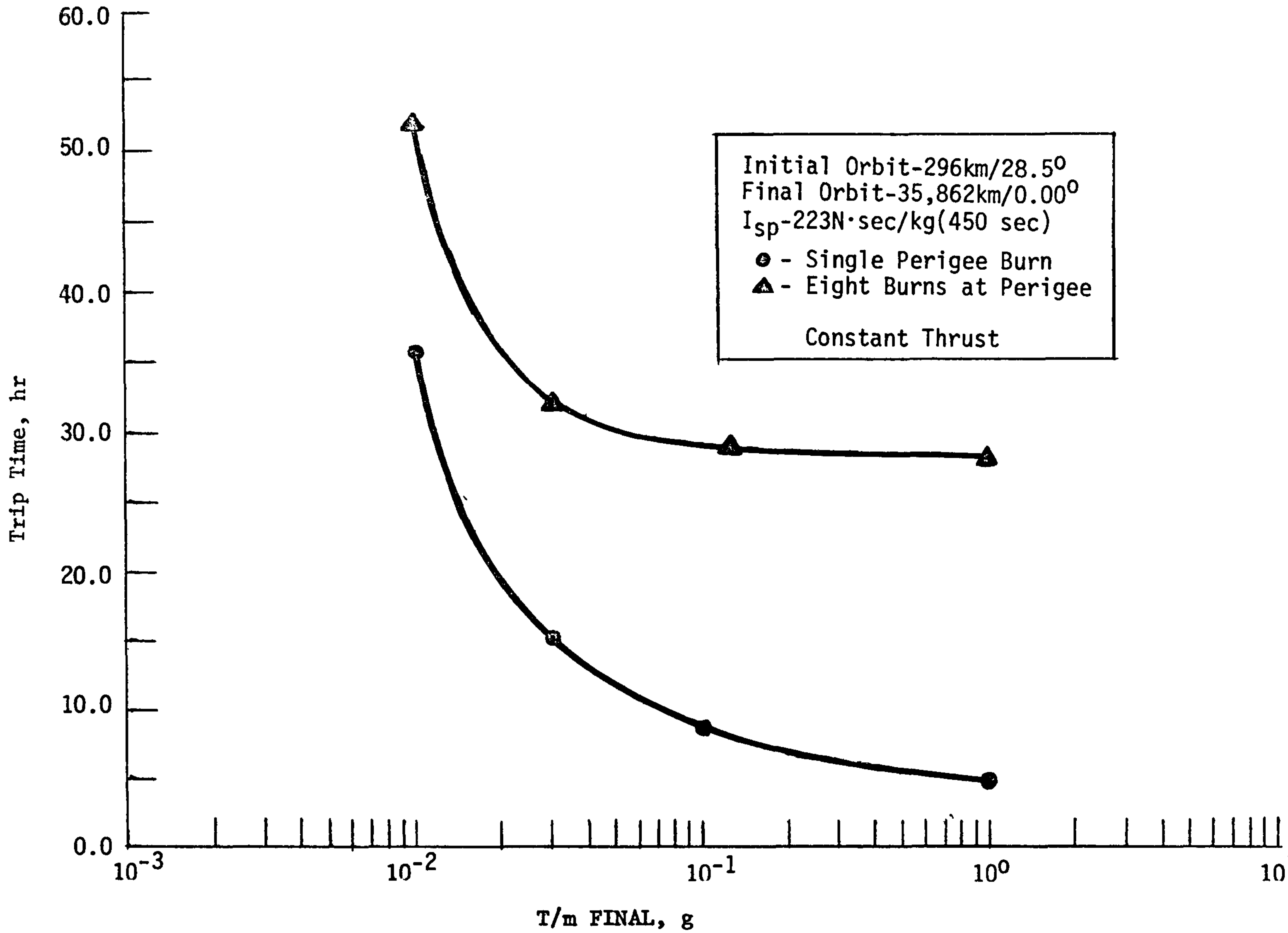


FIGURE II -13 TRIP TIME REQUIREMENTS

by 11 hours the boiloff increases by 34 kg. These increases are associated with a final acceleration of 0.016 g. With the approximate boiloff of 850 kg, the percentage increase is less than 2 percent.

The other area of concern is degradation of electronics by increased dwell time within the Van Allen belts. But the small increase in transfer time is not considered to be a problem and can be solved with adequate shielding.

In Section II-F a more detailed analysis to determine applicable thrust levels is described. This section also discusses how the choice of acceptable T/m_f will affect the mission capture for each engine.

Lengths of each vehicle were calculated for the maximum combined stage/spacecraft mass of 28,000 kg, assuming lighter payloads would then require off-loading of propellants. This maximum mass approach is used since a single length vehicle is considered to be the most realistic scenario. Results from the PP/LSSI Study predicted that most payloads would be mass constrained if an ellipsoidal/toroidal PPS is used, i.e., the maximum mass that could be carried on the STS would be exceeded before the volume available is filled. These results will hold true for this study as the missions identified have smaller payloads than those defined in the previously mentioned study.

Figure II-14 shows the vehicle lengths for the three engine systems. Variations in the vehicle lengths from engine to engine depend mainly on how the engine fits within the inner diameter of the toroidal tank. Profiles showing the three engines embedded within the toroid are displayed in Figure II-15. In each case the largest toroid required (lowest thrust) is shown with its repective engine. A minimum clearance of 5 cm is allowed between the outer layer of the insulation and the retracted nozzle. The RL-10 cannot be embedded completely within the toroid since the bell is too wide, thus the geometry of the nozzle dictates how far the engine extends below the torus.

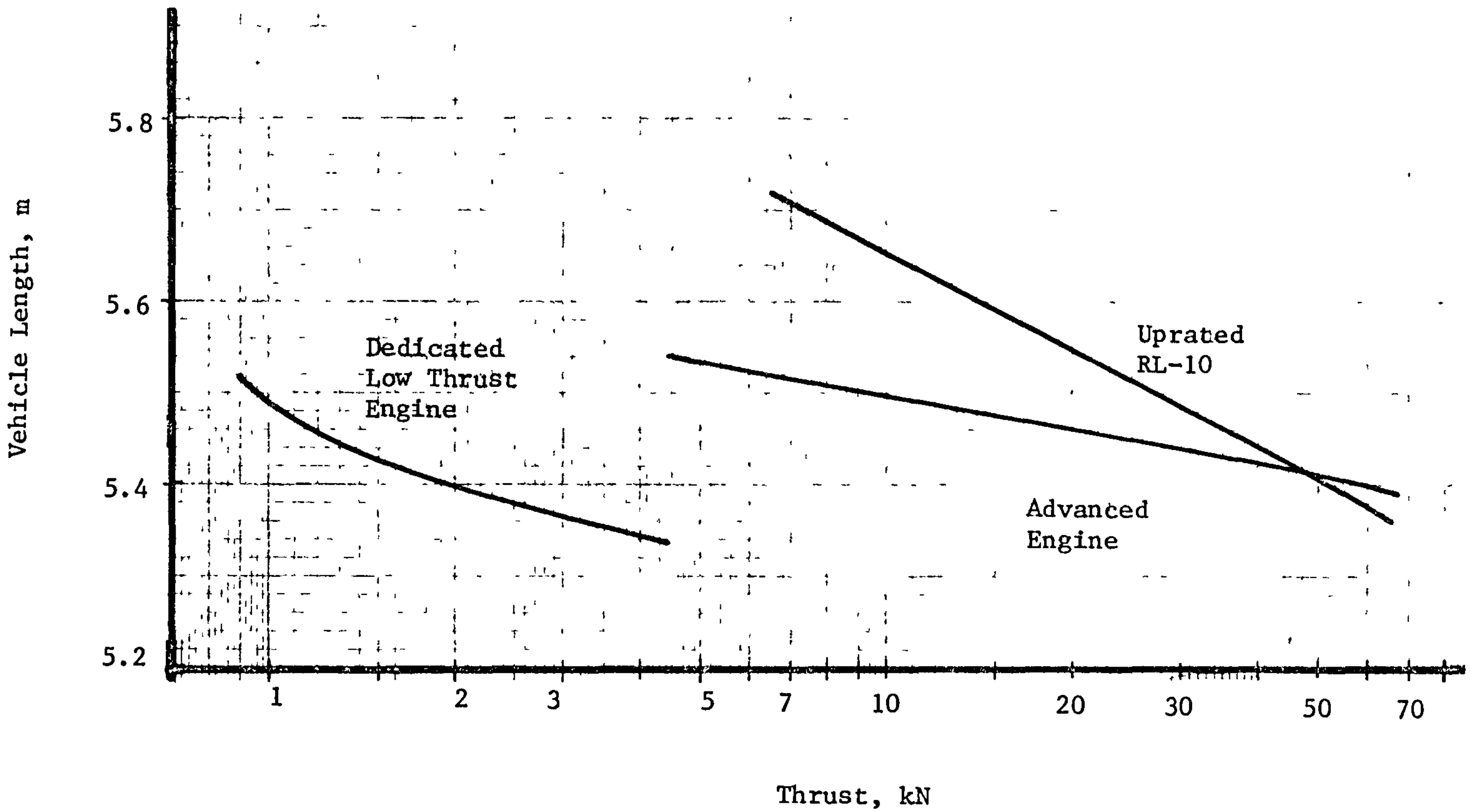
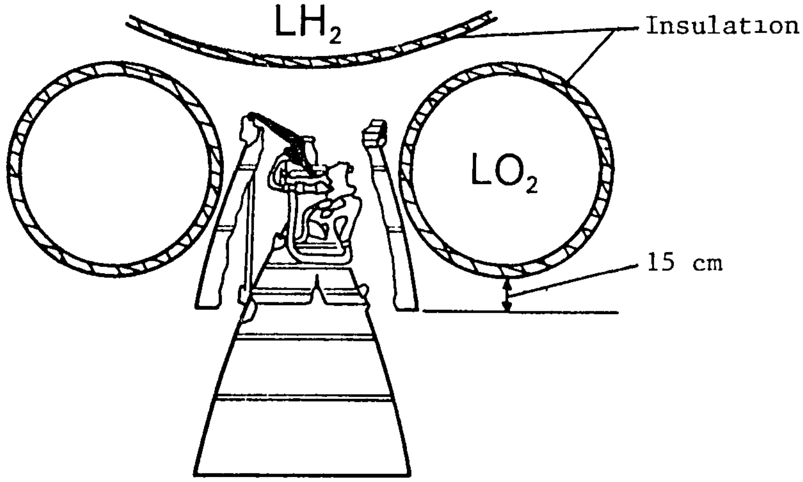
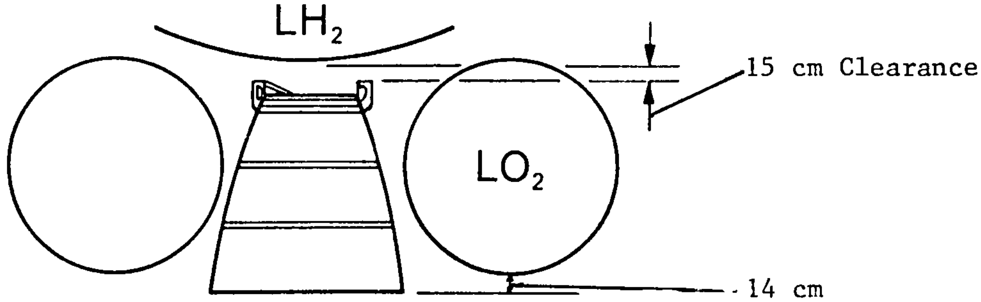


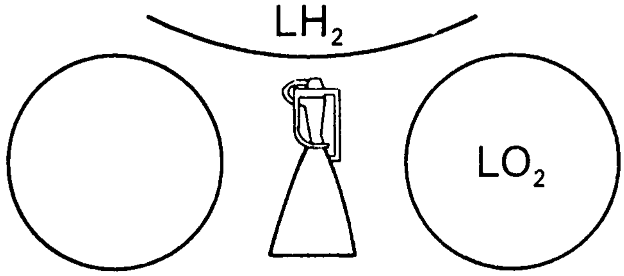
FIGURE II - 14 PPS LENGTH AS A FUNCTION OF THRUST



(a) Uprated RL-10
6700 N Thrust



(b) Advanced Engine,
4450 N Thrust



(c) Dedicated Low
Thrust Engine,
980 N Thrust

FIGURE II -15 ENGINE PLACEMENT

Allowing for a 15 cm clearance between the bottom of the hydrogen tank and the top of the advanced engine causes its nozzle to also extend below the bottom of the torus. However this engine will fit within the inside diameter of the torus. This stage length is calculated from the engine length plus clearance added to the LH_2 tank length plus insulation thickness. In the case of the dedicated low thrust engine, it fits within the toroid inside diameter and is shorter than the height of this tank. Therefore the PPS length for this engine is found by adding the LO_2 tank height plus insulation to the LH_2 tank plus insulation. In Figure II-15 it can be seen that the total system lengths vary by no more than 0.38 m. Thus system length would not be a strong factor in the choice of a PPS.

F. MISSION CAPTURE

Mission capture information determined the compatibility of each engine/PPS combination with the LSS mission catalog. From this work, one can predict the specific missions captured by each engine and the required thrust level, or thrust level range.

Results revealed the following; (1) which mission capture approach should be used in the benefit and cost model, (2) which engine best satisfies the mission catalog requirements.

1) Ground Rules

The following ground rules apply for the mission capture. Each engine/PPS combination was sized for maximum payload delivery to GEO across the full engine thrust range. Payload masses requiring less than the maximum stage capability will be captured by off-loading propellant However, the spacecraft cannot be ballasted to displace the LSS final acceleration range into the mission capture envelope.

An acceleration range was specified in the catalog since no detailed analysis has yet determined the exact design acceleration limit of each spacecraft. The limits catalogued in the mission model are the best estimates available from current literature. Accelerations for a specific LSS span from

the most conservative (lowest acceleration) to the least conservative estimate (highest acceleration). At the lowest value, the mission has a 100% possibility of being captured since the actual LSS design point will be higher. At the highest acceleration level, the possibility of mission capture is 0% since the actual LSS design point is going to be below this value. There is equal probability of finding the actual LSS design point at any value within the specified range. Thus as the thrust level increases, the possibility of capturing a specific mission decreases linearly over the accepted acceleration range. Thrust levels that produce accelerations below a mission's most conservative limit will capture the mission with 100% probability if the engine/PPS combination provides enough payload capacity. Capturing a mission at an acceleration lower than the mission catalog recommends does not significantly alter the engine benefits/cost value. This relaxation of the lower final acceleration value was considered the most realistic approach when considering the number of missions an engine can capture.

2) Approach

If a mission is to be captured by an engine/PPS combination, then two requirements must be met. First, the engine must supply final acceleration within the acceptable range specified for that mission. And second, engine performance at that thrust level must provide the required payload delivery capacity to GEO.

The procedure which determines whether or not the mission can be captured is presented in Figure II-16. Iteration on thrust is the essence of this procedure and is accomplished by the burnout mass versus thrust relationships shown in Table II-9. Both relationships, payload/thrust and burnout-mass/thrust, are derived from sizing the PPS for maximum capacity. These relationships are valid only across the engine thrust range. Selection of an PPS and specific mission begins this procedure (noting the engine thrust level range, the mission payload mass and acceptable final acceleration range).

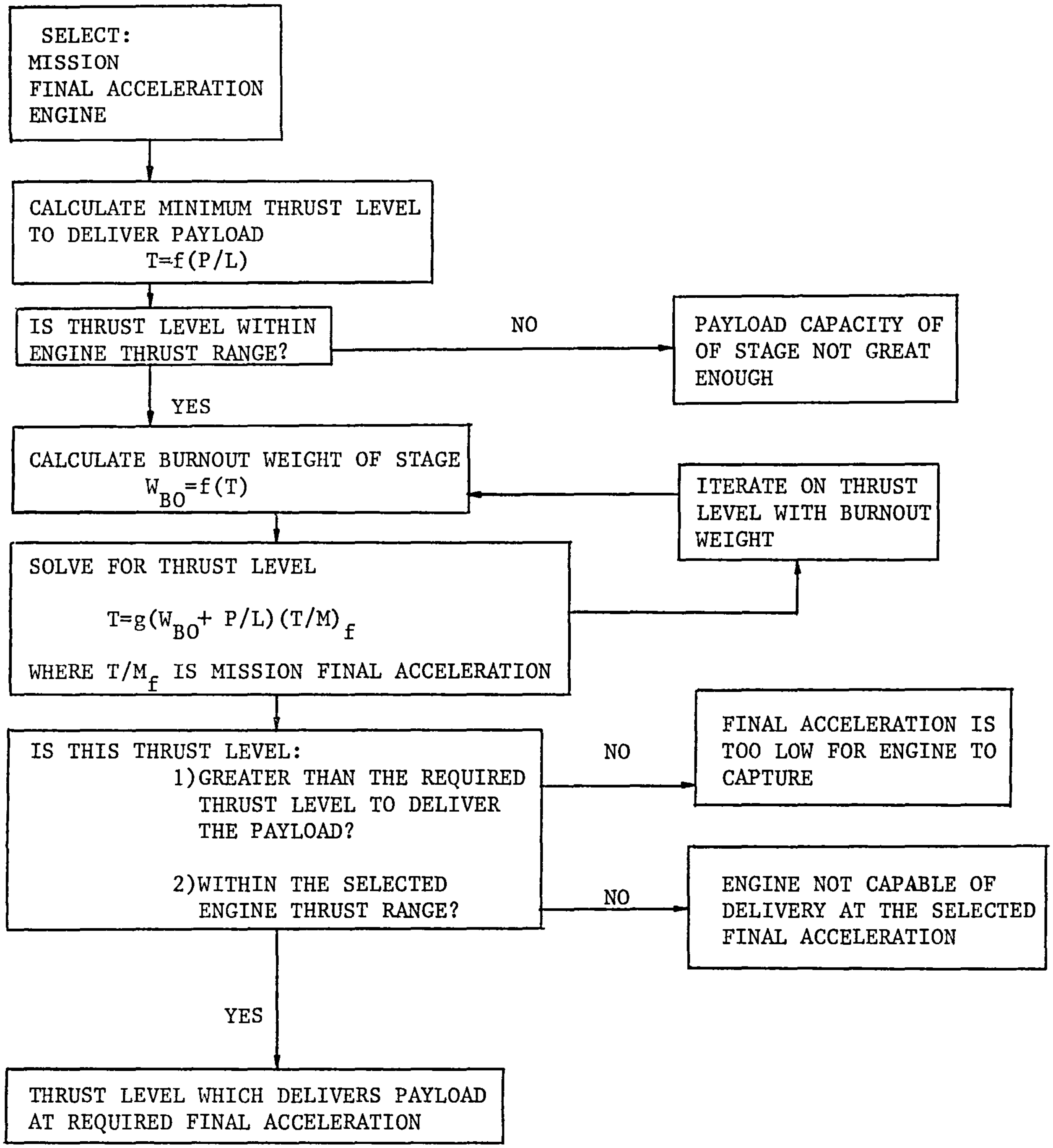


FIGURE II -16 MISSION CAPTURE PROCEDURE

Table II-9 Mission Capture Equations

Engine	Thrust Range, N	Payload/Thrust Relationships	Burnout-Mass/Thrust Relationships
Uprated RL-10	6672 to 66720	$P/L(kg) = 3705.8(T)^{0.067}$	$W_{Bo}(kg) = 3079.1(T)^{-0.0075}$
Advanced	4448 to 66720	$P/L(kg) = 4666.7(T)^{0.051}$	$W_{Bo}(kg) = 2980.0(T)^{-0.0062}$
Dedicated Low Thrust	890 to 4448	$P/L(kg) = 4217.6(T)^{0.072}$	$W_{Bo}(kg) = 2876.3(T)^{-0.0087}$

A minimum thrust level that will deliver the payload is calculated. If this thrust level is within the engine thrust level range then the expressions containing burnout mass and thrust level are solved simultaneously. Thrust levels derived from this procedure will capture the mission only if the engine performance equals or exceeds that required to deliver the payload to GEO. Applying the procedure at both endpoints of the final acceleration range produces a thrust range for a specific engine which will capture that mission.

3) Predictions From Mission Capture Equations

Actual mission capture matched each PPS to the mission requirements for both mass deliverable and payload acceleration limits. The mission capture of the uprated RL-10, advanced engine, and dedicated low thrust engine are presented in Table II-10.

The engine/mission capture can be one of three possible states. These three states are fully captured, partially captured, or none captured. Fully captured means the engine can provide an acceleration less than or equal to the most conservative estimate which delivers the necessary payload mass. The capture states of each mission are discussed in the following engine sections.

TABLE II-10 ENGINE/PPS PERFORMANCE AND MISSIONS CAPTURED

Engine	Thrust, N (lbf)	GEO Deliverable Payload Mass, kg (lbm)	Possible Number of Missions Captured
Dedicated Low Thrust	890-4450 (200-1000)	6810-7690 (15,000-16,950)	25
Advanced	4450-66, 700 (1000-15,000)	7180-8240 (15,820-18,179)	18
Uprated Rl-10	6670-66,700 (1500-15,000)	6660-7760 (14,680-16,450)	15

a) Uprated RL-10

Lowest thrust levels available with the uprated RL-10 allowed full capture of only five missions. Ten other missions are partially captured while thirteen exceed the deliverable requirements of this engine/PPS. Table II-11 lists the missions and their compatability with the uprated RL-10.

TABLE II-11 UPRATED RL-10 PPS MISSION CAPTURE

Mission Number	State
1, 3, 6, 8, 13	Can be fully captured at some engine thrust level
4, 5, 7, 9, 10, 16, 18, 22, 24, 25	Partially captured
12, 14, 15, 17, 19, 20, 21, 23, 26, 27, 28	None captured, payload exceeds delivery capability of PPS/engine combination
2, 11	None captured, uprated RL-10 cannot provide thrust low enough to capture these missions.

b) Advanced Engine

Increased performance characteristics of the advanced engine allows for more compatability with the mission model than the uprated RL-10. This can be seen by the greater number of missions captured for the advanced engine in Table II-10. Only three missions exceed the payload capacity of the advanced engine but eight can be fully captured. Nine missions cannot be delivered at the 4450 N thrust level because the required delivery capability is too low. Two characteristics which improved the mission capture for the advanced engine over the uprated RL-10 engine is a lower minumum thrust level, 4450 vs 6670 N, and a higher Isp. Results of these two differences can be seen graphically in Figure II-12 (presented in Section II-E) where for example the acceleration range for mission 17 falls completely within the thrust/payload envelope for the advanced engine. Although this spacecraft mass is within delivery capabilities of the uprated RL-10, the thrust required to deliver mission 17 results in a T/m_f too high for the structure to withstand. The higher Isp of the advanced engine delivers the required mass at a lower thrust level.

Table II-12, similar to the one in the previous sub-section, lists the states of the 28 missions identifed in the catalog.

TABLE II-12 ADVANCED PPS MISSION CAPTURE

Mission Number	State
1, 3, 5, 6, 8, 13, 17, 27	Fully captured
4, 7, 9, 10, 11, 12, 16, 18, 20, 21, 22, 23, 24, 25, 26, 28	Partially captured
14, 15, 19	None captured - exceeds payload capacity
2	None captured - accleration required is too low

c) Dedicated Low Thrust Engine

Reasons for an improved mission model capture using the advanced engine over the RL-10 are also responsible for a further improvement for the dedicated low thrust PPS. Only three missions are not deliverable with this engine, 14, 15, and 19, but both spacecraft have masses larger than any engine

delivery capabilities with thrust levels acceptable to the structures. This engine has enough performance to deliver many missions even at T/m_f below the least conservative estimate of acceleration. Lower thrust levels than recommended will increase transfer times along with all the attendant problems but as discussed earlier the effects are not significant. Capture performance of the dedicated low thrust PPS is shown in Taole II-13.

TABLE II-13 DEDICATED LOW THRUST PPS MISSION CAPTURE

Mission Number	State
1, 3, 5, 6, 8	Fully captured across full thrust range of the dedicated low thrust engine.
4, 7, 9, 10, 12, 13, 16, 17, 18, 20, 21, 22, 23, 24, 25, 26, 27, 28	Fully captured at some thrust level within the dedicated low thrust engine capability
2, 11	Partially captured
14, 15, 19	None captured, exceed payload capability.

From the mission model captures of each engine, the dedicated low thrust PPS was seen to have the most compatible T/m_f and payload delivery capabilities. This result is partially due to the large number of missions that do not come close to the maximum payload capabilities of any engine. For these missions only an upper limit on acceleration needs to be satisfied. The tnrust level which captures this group of missions will produce an acceleration equal to the lowest value of the "most conservative acceleration" for the group of missions. Obviously the lower the thrust available the better the probability of capture.

Mission capture tradeoffs were considered for each engine. The choice of the most appropriate single thrust level included weighting each missions overall importance, capture index, and number of flights. Methods for mission capture tradeoffs will be evaluated in Task II.

Since the mission model was developed independently from the PPS sizing the only consideration was to include spacecraft that cover the full range of requirements. These mission catalogs can be seen to fulfill the stipulation, thus no adjustments were considered appropriate. Task II analyses included the weighting of missions to allow mission prioritization.

III. TASK II - BENEFIT VERSUS COST ANALYSIS MODEL DEVELOPMENT

This task determined the life cycle cost (LCC) parameters and approach used for the benefit model development.

A. COST ALGORITHM

1) Define LCC Parameters

LCC parameters which describe the cost incurred for transferring spacecraft from Earth to GEO are shown in Figure III-1. These cost categories will act as a guide in the development of cost estimating relationships (CERs). Each category, with the exception of launch and deployment operations, has two CERs. One relates research, development, test and engineering (RDT&E) costs to design parameters and the other CER is first unit costs. The RDT&E costs included all those incurred during concept formulation, validation, and full scale development phases of a program. Included are costs of feasibility studies, preliminary design, engineering design/development, fabrication, assembly and checkout of prototypes and test units, initial system evaluation, and associated documentation costs.

2) Cost Data Base Generation

The accuracy of a cost estimating methodology depends primarily on the extent, usefulness, and appropriateness of the data from which it is built. To ensure the most accurate data base possible, three data searches were undertaken, 1) analysis of past programs 2) literature search 3) and vendor contact.

a) Analysis of Past Programs

Past in-house programs were examined for their applicability to this study and their actual costs. Programs analyzed include Viking, Titan, External Tank, Transtage, Scatha and RCS Tanks,

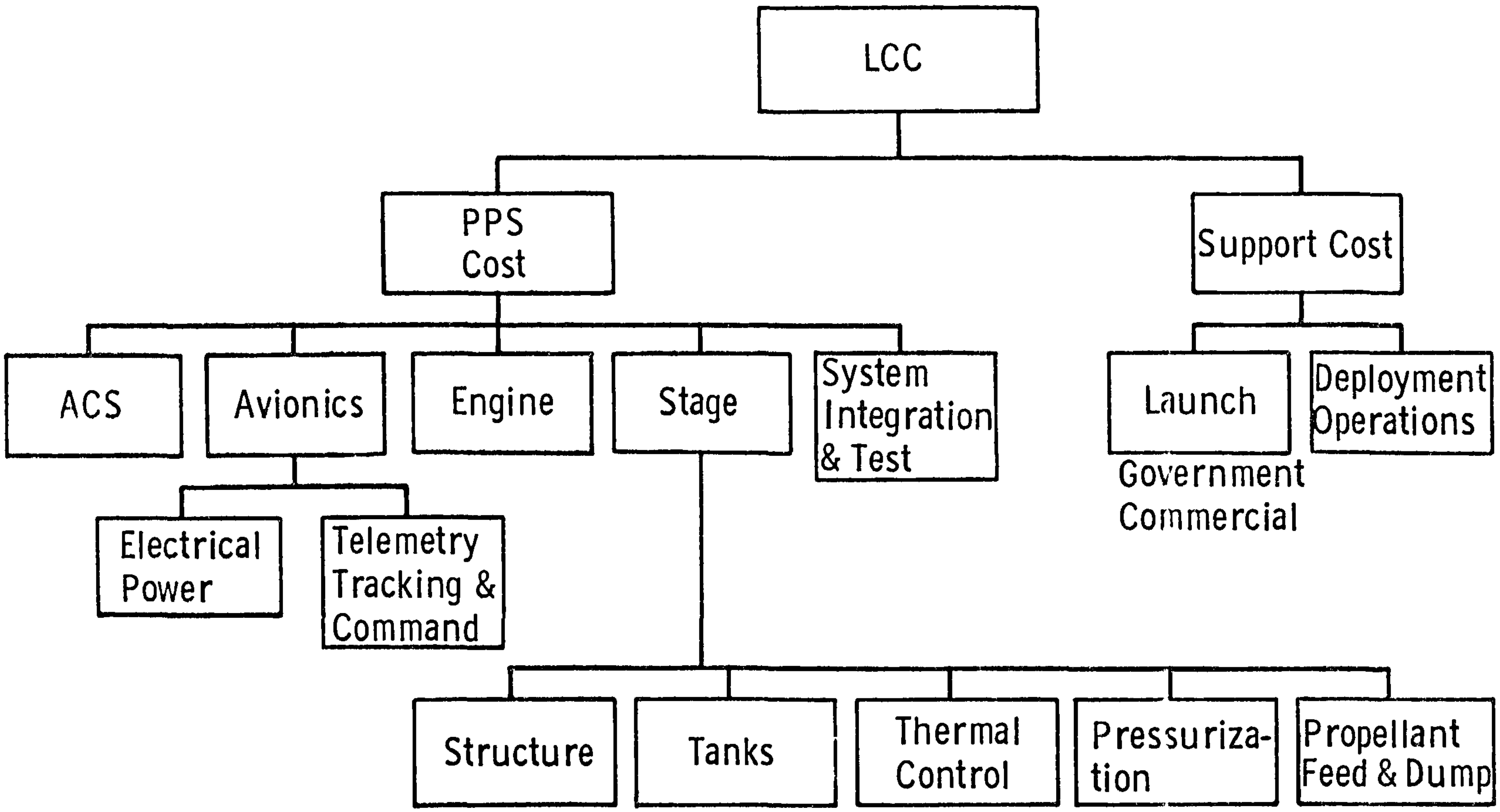


FIGURE III-1 Cost Breakdown by Subsystem

all hardware build projects. Study projects which included detailed bottoms-up cost estimates were also examined. These included Space Tug, Teleoperator Maneuvering System, MX Stage IV Propellant Storage Assembly, and Orbital Transfer Vehicles.

b) Literature Search

The literature search was intended to gather all documented costing techniques applicable to this study to provide insight and guidance for the development of the model. A search was also conducted for published cost data directly relating to the type of system to be priced. As expected, very little actual cost data was identified. However, the USAF Space Division Unmanned Spacecraft Cost Model was of considerable aid in formulating the model.

c) Vendor Contact

For hardware items which Martin Marietta has not previously built and could not be located through literature searches, potential vendors were contacted to supply rough order of magnitude cost estimates. This activity proved very beneficial in filling data gaps. Companies which supply valves, filters, regulators and propellant lines were contacted regarding costs for each.

3) Cost Estimating Relationships (CER)

Once the raw cost data had been gathered, validation and cost estimating relationship (CER) formulation activities were undertaken. All of the data was validated to ensure that it was relative and pertinent to this study. During the validation phase, the engineer who supervised the design task was interviewed. This permitted the exploration of erratic data and often provided for additional data. Whenever a unique project problem was identified the cost impact was determined and the data point was normalized. Additionally, all cost data was escalated using a composite index to 1982 values.

The cost data was then transformed into cost estimating relationships using established statistical regression procedures. An experimental equation provided the best statistical fit for the majority of the relationships. Each equation yields a cost in FY82 dollars expressed in thousands. The following sections provides a description of each subsystem and their associated cost estimating relationships.

a) Stage Costs

Telemetry, Tracking and Command - Performs one or more of the following functions: measures important spacecraft platform conditions; processes this information and mission data; stores and transmits data to ground, receives and processes commands from ground and initiates their execution; and provides a tracking capability. Typical equipment includes analog/digital converters, coders, digital electronics (digital storage units, command distribution units, programmers, etc) or computers, signal conditioners, format control units, transmitters, antennas, receivers, decoders, switching relays, tape recorders, amplifiers and clocks.

RDTE \$ = 1188.68 + 54.81 (Telemetry Tracking and Command Weight, lbs).

Unit \$ = 51.34 + 36.94 (Telemetry Tracking and Command Weight, lbs).

Attitude Control System (ACS) - Maintains the spacecraft in the required orbit. It also maintains the correct attitude and direction of determined axes within that orbit. This is achieved by sensing the spacecraft attitude at all times and making necessary adjustments. The ACS subsystems consist of two functional categories of equipment. The first category is attitude determination equipment, typically includes sun sensors, horizon scanners or sensors, star sensors, control and gyro electronic. The second category is attitude and reaction control which typically includes reaction control

nozzles, fuel lines, valves, fuel tanks, nutation dampers, wobble dampers, and gravity booms.

$$\text{RDT\&E } \$ = 1494.78 + 98.61 \text{ (Altitude Control Dry Weight, lbs)}^{0.81}$$

$$\text{Unit } \$ = 17.59 \text{ (Altitude Control Dry Weight, lbs)}^{0.69}$$

Electrical Power Supply for PPS - Stores, regulates, and distributes all electrical energy to and between spacecraft components. Equipment includes batteries, regulators, converters, power distribution units and wire harnesses.

$$\text{RDT\&E } \$ = 2648.8 + 0.031 \text{ (Electrical Power Supply X Power Level, lbs - watts)}^{0.97}$$

$$\text{Unit } \$ = 66.72 \text{ (Electrical Power Supply X Power Level, lbs - watts)}^{0.29}$$

Thermal Control - Maintains the temperature of the stage and engine within allowable limits. Thermal control includes passive methods (paint, insulation) and/or active methods (radiators, heaters, temperature sensors and heat pipes).

$$\text{RDT\&E} = 251.62 + 29.46 \text{ (Thermal Control Weight, lbs)}^{0.66}$$

$$\text{Unit } \$ = 4.25 \text{ (Thermal Control Weight, lbs)}^{0.65}$$

Tanks - Contain fuel and oxidizer for the PPS. Equipment includes barrel section, domes, propellant acquisition devices and bubble filters.

LIQUID OXYGEN TANK

$$\text{RDT\&E } \$ = 9674.5 \text{ (}LO_2\text{ Tank Weight, lbs)}^{0.13}$$

$$\text{Unit } \$ = 15.8 \text{ (}LO_2\text{ Tank Weight, lbs)}^{0.68}$$

LIQUID HYDROGEN TANK

RDT&E $ = 3869.8 (LH_2 Tank Weight, lbs)$^{0.13}$

Unit $ = 7.91 (LH_2 Tank Weight, lbs)$^{0.68}$

Propellant Feed and Dump System - Provides the capability of transferring propellants from their tanks to the engine or to space during a Shuttle abort. Equipment includes feedlines, burst discs, and valves.

RDT&E $ = 1382.0 (Feed and Dump Weight, lbs)$^{0.21}$

Unit $ = 114.0 + 0.08 (Feed and Dump Weight, lbs)

Structure - The structural support which acts as the primary support of the stage and thrust structure.

RDT&E $ = 754.9 + 70.8 (Structure Weight, lbs)$^{0.66}$

Unit $ = 2.51 (Structure Weight, lbs.)$^{0.96}$

Pressurization System - Provides PPS with the required pressure level to maintain performance. Equipment includes lines, tanks, filters, regulators, valves and necessary hardware for a thermal subcooler in the LH_2 tank.

RDT&E $ = 3289.0 (Pressurization System Weight, lbs)$^{0.21}$

Unit $ = 157.0 + 0.42 (Pressurization System Weight, lbs)$^{0.77}$

System Integration and Test - Includes those cost areas which cannot be related to any specific subsystem. Included in this area are program management, systems engineering, systems test and evaluation, acceptance test, reliability/quality assurance and configuration and data mangement.

RDT&E COSTS

System Engineering & Management $ = 0.25 (Total Hardware RDT&E Costs)

Systems Test $ = 0.45 (Total Hardware RDT&E Costs)

UNIT COSTS

Systems management, Integration, & Test $ = 0.30 (Total Hardware Unit Cost)

b) Support Costs

Launch - The cost of placing the stage and LSS into LEO. User charge for the Shuttle includes all consumables, launch operations, and applicable amortizations. Commercial users are assessed a $55.7M charge for a dedicated flight while government users are charged $31.3M. This study assumes that all Shuttle flights are dedicated.

Deployment Operations - The cost for monitoring the mission while transferring the LSS from LEO to GEO. Both personnel and equipment usage costs are included. The deployment operations cost represents an average cost per hour for deploying a spacecraft less any special costs due to the needs of the payload.

Deployment Operations Cost = 1.43 (Hours of Ground Control Operation Time)

Total hours of ground control operation time is the sum of LEO checkout time for the payload, 42 hours, and the triptime to GEO. Triptime is a function of final spacecraft acceleration. Thrust level, PPS burnout mass, and payload mass combine to yield a final acceleration. For an eight perigee burn, one apogee burn orbit transfer strategy, the triptime as a function of final acceleration is displayed in Figure II-13. This data represents a spacecraft transfer

time from LEO @ 296 km, 28.5° inclination to GEO @ 35,889 km, 0° inclination for an eight perigee, one apogee burn scenario. This function must be expressed mathematically to facilitate its use in the benefit and cost model. Exponential and power series curve fits resulted with correlation coefficients of 0.68 indicating very poor modeling. The chosen curve fit employs four linear segments which are within five percent of the functions in Figure II-13. Table III-1 displays each final acceleration range and the accompanying triptime versus final acceleration expression for the eight perigee burn scenario.

TABLE III-1-MATHMATICAL EXPRESSION OF TRIPTIME AS A FUNCTION OF FINAL ACCELERATION

FINAL ACCELERATION RANGE g	TRIPTIME [HR] = f (FINAL ACCELERATION [g])
$g < 0.012$	$Y = -5000.0\ g + 100.33$
$0.012 \leq g < 0.017$	$Y = -1288.5\ g + 55.9$
$0.017 \leq g < 0.03$	$Y = -345.6\ g + 39.9$
$0.03 \leq g$	$Y = -1.74\ g + 29.5$

NOTE: EIGHT PERIGEE BURN, ONE APOGEE BURN STRATEGY

4) Life Cycle Cost Flow Chart

These CERs were used in the benefits and costs program to determine total RTD&E costs and first unit costs. A flow diagram explaining the calculation process is shown in Figure III-2.

Production costs incorporated a learning curve for the primary propulsion system and liquid rocket engines. A ninety percent learning curve applies to the primary propulsion system. The advanced engine and the dedicated low thrust engine have a ninety-two percent learning curve. Since the improvement of the RL-10 engine does not involve a major redesign, the uprated RL-10 has no learning curve.

B. BENEFIT ALGORITHM

The benefit section of the model follows a weighted criteria rating approach. A list of benefit criteria was established along with gudelines for their evaluation. These criteria can be rated as to their relative weights within the total system worth. The sum of all criteria multiplied by their weighting factor for each stage represents the PPS Benefit. The maximum value for the benefit value is 10.0.

Since mission capture was expected to have a major impact on the benefit associated with each stage, a separate methodology was developed for its rating. Because there was doubt as to whether all missions would actually occur, the model can handle each mission on a likelihood basis. This prevents a stage from receiving a low rating due to its inability to capture a mission that has a low likelihood of occurrence. This procedure is open-ended to enable the addition and/or deletion of missions. The major driver in the mission capture analysis was the thrust range each LSS is expected to be able to tolerate.

RDT AND E COST

TELEMETRY, TRACKING, AND CONTROL
+ ATTITUDE CONTROL
+ ELECTRICAL POWER SUPPLY
+ LH_2 PROPELLANT TANK
+ LO_2 PROPELLANT TANK
+ STAGE STRUCTURE
+ FEED AND DUMP SYSTEM
+ PRESSURIZATION SYSTEM
+ THERMAL CONTROL - PASSIVE

= HARDWARE RDT&E COST
+ SYSTEMS ENGINEERING MANAGEMENT
+ SYSTEMS TEST

= TOTAL RDT AND E COST

SUPPORT COST

GOVERNMENT LAUNCH COST
+ COMMERCIAL LAUNCH COST

= TOTAL LAUNCH COST
+ DEPLOYMENT OPERATIONS

= SUPPORT COST

PRODUCTION COST

TELEMETRY, TRACKING, AND CONTROL
+ ATTITUDE CONTROL
+ ELECTRICAL POWER SUPPLY
+ LH_2 PROPELLANT TANK
+ LO_2 PROPELLANT TANK
+ STAGE STRUCTURE
+ FEED AND DUMP SYSTEM
+ PRESSURIZATION SYSTEM
+ THERMAL CONTROL - PASSIVE

= HARDWARE PRODUCTION COST
+ SYSTEM MANAGEMENT, INTEGRATION, & TEST

= FIRST UNIT PRODUCTION COST

TOTAL PRODUCTION COST = (FIRST UNIT COST) $\sum_{i=1}^{n} X_i^{(Y)}$

WHERE n EQUALS TOTAL PRODUCTION QUANTITY

Y is the learning exponential

ENGINE COST

TOTAL RDT AND E COST
+ SUPPORT COST
+ TOTAL PRODUCTION COST
+ ENGINE COST

= LIFE CYCLE COST

FIGURE III-2 COST MODEL FLOW CHART

To determine the most beneficial thrust level for operating an engine the benefit algorithm iterates on thrust level. After selecting a thrust level increment, the computerized benefit algorithm calculates the mission capture rating, stage length rating, and benefit for each thrust value.

1) Benefit Criteria

The following is a list of the benefit criteria and their rating scales which have been identified and what each represents.

a) Mission Capture

Mission Capture is the ability of the stage to satisfy the deployment constraints of each identified LSS. Prime factors to be included are thrust level, performance, and likelihood of each mission. The likelihood factor is included to prevent missions with a low probability of occurrence from driving the model. The requirements of the model are 1) as the likelihood of a specific mission decreases, the mission capture factor increases and 2) as the probability of a LSS to support a given thrust level increases, the mission capture increases. The resulting model is present in the following equation.

$$\Omega = 10 \times \left[1 + \left(\frac{\Sigma \beta (\psi - 1)(\alpha)}{\Sigma \alpha} \right) \right]$$

Where Ω = mission capture rating

β = Likelihood of mission occurrence with a value from 0 to 1

ψ = mission captures index with a value from 0 to 1, a linear function of thrust

α = number of stages required to capture a mission model.

The benefits/costs model calculates the mission capture rating by generating ψ for each mission and from the inputs of α and β.

b) Engine Reliability

Reliability is the ability of the engine to successfully complete each mission.

Each Rating Point = .003

Rating		Reliability
10	=	1.000
7	=	0.991
4	=	0.982
0	=	0.970

c) Engine Technical Risk

Engine Technical Risk is the confidence level that the engine can be built to the defined specifications.

Rating	Technical Risk
10 =	off the shelf hardware
7 =	minor modification to an existing design, e.g., redesign pumps for mixture ratio change,
5 =	major modification to an existing design
3 =	new design, state-of-the-art
0 =	theorized new technology

d) Growth Potential

Growth Potential is the ability of the engine to be altered at a future time to increase its performance or applications.

For 10% more performance, effort required is

10	=	none
7	=	minor alteration
3	=	major alteration
0	=	total redesign

e) Length of Development

Length of Development is the time expected to design, develop, test and evaluate the engine.

Each Rating Point = 1.0 year

Rating		Length of Development
10	=	2 years
7	=	5 years
5	=	7 years
3	=	9 years
0	=	12 years

f) Engine Technical Desirability

Engine Technical Desirability is the benefits gained by any new technologies that may be developed during the engine evolution.

Rating		Technical Desirability
10	=	technological breakthrough
5	=	new use of existing technology
0	=	none

g) <u>Stage Reliability</u>

Stage Reliability is the ability of the stage to successfully complete the mission.

Each Rating Point = 0.003

Rating		Stage Reliability
10	=	1.000
7	=	0.991
4	=	0.982
0	=	0.970

h) <u>Stage Length</u>

Size of the physical length of the stage. Stage length varies with thrust level.

Each Rating Point = 0.3 meters

Rating		Length
10	=	4.6 meters
7	=	5.5 meters
5	=	7.1 meters
3	=	6.7 meters
0	=	7.6 meters

i) <u>Fabricability</u>

Fabricability is the ability to incorporate existing fabrication techniques into the stage production phase.

Rating		Fabricability
10	=	existing manufacturing techniques
7	=	minor modification of manufacturing techniques
5	=	major modification of manufacturing techniques
3	=	new state-of-the-art manufacturing techniques
0	=	new theorized manufacturing techniques

j) Repairability

Repairability (in orbit) assumes the existence of a space service vehicle and the ability to remove and replace a failed component without returning the stage to Earth. The repairability percentage refers to the ratio of components that can be repaired in orbit to the total number of components.

Rating		Repairability
10	=	100%
7	=	70%
5	=	50%
3	=	30%
0	=	0%

C. EXAMPLE BENEFIT CALCULATION

This section provides an example benefit calculation using fictitious values for the mission model, rating and weighting factors. The intent of the section is to give an in-depth understanding of the benefit algorithm. Fictitious values for the mission model, rating, and weighting factors are used to reduce and simplify sample calculation.

First, the mission capture rating will be calculated. The equation that models mission capture rating is shown below.

$$\Omega = 10 \times \left[1 + \left(\frac{\Sigma \beta (\psi - 1)(\alpha)}{\Sigma \alpha} \right) \right]$$

Ω = likelihood of mission occurrence (0 to 1)
ψ = mission capture index (0 to 1)
α = number of stages required to capture mission

Values of β, ψ, and α along with a sample calculation for mission capture rating are shown in Table III-2. The sum of the products of β, $(\psi-1)$, and α are displayed in the table.

Mission capture index is a function of thrust level as discussed in section III-B-1. As thrust level increases, mission capture index decreases for all missions.

Similarly, as thrust level increases the stage length decreases. These two parameters have adverse effects on the benefit of a PPS since higher thrusts decrease the mission capture rating but increase the stage length rating.

The capture rating calculation continues following Table III-2. Mission capture rating for this sample mission model is 5.346.

This capture rating is placed in the benefit evaluation, Table III-3, where all benefit criteria are listed with rating and weighting factor values. The benefit of this sample engine at a sample thrust level is 4.8346.

After the example benefit calculation is studied, the reader should have insight into the benefit algorithm and how mission capture rating and stage length varies with thrust level.

D. BENEFITS AND COST ANALYSIS MODEL

Using the previous two algorithms, a benefits and costs analysis model was developed. The model named RACE (Rating And Cost of Engine) is written in Fortran IV. When supplied with a mission model and primary propulsion system information, RACE calculates the PPS RDT&E and first unit costs. RACE iterates on thrust level across the thrust range of interest. At each thrust level, the model calculates the mission capture rating, benefit, and LCC. This information is vital in determining the most beneficial and least expensive thrust level

TABLE III-2 - SAMPLE MISSION CAPTURE RATING

Mission	Likelihood β	Mission Capture Index	$\psi - 1$	$\beta(\psi - 1)$	Flights α	$\alpha\beta(\psi - 1)$
1	1.0	0.8	-0.2	-0.20	4	-0.80
2	0.8	1.0	0.0	0.00	3	0.00
3	0.5	0.8	-0.2	-0.10	1	-0.10
4	0.8	0.9	-0.1	-0.08	1	-0.08
5	1.0	0.0	-1.0	-1.0	6	-6.00
				Σ	15	-6.98

$$\frac{\Sigma\alpha\beta(\psi - 1)}{\Sigma\alpha} = \frac{-6.98}{15.0} = 0.4653 \text{ (D)}$$

$$D+1 = 0.5347 \text{ (E)} \qquad \Omega = 10 \times E = 5.347$$

at which to operate. If the program is supplied with information about another PPS, then the most beneficial and cost effective PPS can be selected based on benefits and cost comparison. A program listing of RACE, its logic flow chart, and input format code appear in Appendix B.

E. BENEFITS AND COSTS OF ADVANCED PPS USING RACE

To further the understanding of RACE an example will be presented using the advanced PPS. Detailed input information for this PPS and the NASA/Commercial LSS mission catalog will be given.

Inputs to RACE include the following:

- number of missions
- mission acceleration range
- payload mass
- mission probability
- stages required per mission

TABLE III-3 - BENEFIT EVALUATION

Engine Example

Thrust Level Example

Criteria	Rating	Weighting Factor	Rating X Weighting Factor
Mission Capture	5.346	10	53.46
Engine Reliability	9.00	10	90.0
Engine Tech. Risk	7.0	10	70.0
Growth Potential	2.0	10	20.0
Length of Development	5.0	10	50.0
Technical Desirability	7.0	10	70.0
Stage Reliability	4.0	10	40.0
Stage Length	3.0	10	30.0
Fabricability	6.0	10	60.0
Repairability (In Orbit)	0.0	10	0.0
		100	483.46

Benefit = 483.46/100 = 4.8346

designation of government or commercial mission
initial, final and PPS thrust level increment
liquid rocket engine RDT&E and first unit cost
propulsion subsystem masses
benefit criteria ratings and weighting factors

1) Race Input for NASA/Commercial Mission Catalog

As shown in the previous section, the mission catalog must be quite detailed for input into the benefit and cost analysis model. Each mission has seven required inputs. The order of these inputs are as follows:

1. Mission Number
2. Payload Mass (kg)
3. Most Conservative Acceleration (g)
4. Least Conservative Acceleration (g)
5. Mission Probability
6. Number of Stages
7. Government or Commercial Mission

This information for the NASA/Commercial LSS mission catalog is presented in Table III-4. Mission identification numbers refers to the NASA/Commercial catalog in Section II-C. The GSO communication platform (identification number 6) consists of six spacecraft each with a payload mass of 8200 kg (18,100 lbm). However each spacecraft was divided into two 4100 kg (9,500 lbm) sections to promote mission capture. Thus the number of PPS required to capture Mission 6 is twelve.

2) Benefit Weighting and Rating Factors

The benefit algorithm employs a weighted criteria rating approach. Criteria are weighted the same for comparing propulsion systems. To compare propulsion systems where one propulsion system

criteria were weighted differently than the others would show a bias. However, each propulsion system has a unique criteria rating that characterizes the system.

The benefit inputs required by RACE consist of 10 criteria weighting factors, and chosen criteria ratings. Two criteria ratings, mission capture and stage length, are calculated by RACE. These calculations can be overridden with any rating value greater than 0.0 for those specific criteria. Rating and weighting factors for the advanced PPS are shown in Table III-5.

TABLE III-4 NASA/COMMERCIAL MISSION INFORMATION

MISSION NUMBER	PAYLOAD MASS (kg)	MOST CONSERVATIVE ACCELERATION (g)	LEAST CONSERVATIVE ACCELERATION (g)	PROBABILITY	NUMBER OF STAGES	TYPE* OF STAGE
1.0	2400.0	.999	1.001	1.00	1.0	1.0
2.0	1600.0	.020	.050	.95	1.0	1.0
3.0	4540.0	.200	1.000	1.00	1.0	2.0
4.0	4550.0	.020	.100	1.00	8.0	1.0
5.0	3090.0	.100	.200	.50	2.0	1.0
6.0	4100.0	.100	.500	1.00	12.0	2.0
7.0	5900.0	.050	.200	.20	2.0	1.0
8.0	4540.0	.150	.400	.10	2.0	2.0
9.0	3030.0	.050	.200	.85	2.0	2.0
10.0	6800.0	.010	.100	.20	8.0	2.0
11.0	3100.0	.010	.100	1.00	16.0	1.0
12.0	7500.0	.050	.200	.70	4.0	1.0
13.0	7260.0	.249	.251	.80	1.0	2.0
14.0	8200.0	.050	.350	.50	4.0	1.0
15.0	8200.0	.050	.350	.30	2.0	1.0
16.0	7160.0	.050	.350	.50	2.0	2.0

* 1- Government Application
2- Commercial Application

3) Propulsion Subsystem Masses

First unit costs and RDT&E costs are functions of propulsion subsystem masses except for the elctrical power subsystem. Costs of this subsystem are functions the power-mass product. Masses for each propulsion system being considered were determined during Task I. Propulsion system masses were required to determine stage sizing. Table III-6 presents the propulsion subsystem masses for the advanced PPS example case. These subsystem masses in kg are inputed into RACE for cost calculations. RACE will convert propulsion subsystem masses in kg to lb_f as required by the cost algorithm.

4) Liquid Rocket Engine Information

Completing the example RACE input, liquid rocket engine costs and thrust level range are added to the PPS information. Specifically, RDT&E and First Unit Cost of the liquid rocket engine will complete the cost calculation. Cost values of RDT&E and first unit for the advanced PPS are $270 million and $2.8 million respectively.

As mentioned previously RACE iterates on thrust level. Therefore, the only addition information needed for the example case is the initial thrust level, final thrust level, and thrust increment. These thrust values are the advanced PPS engine thrust range endpoints, 4450 N and 66,700 N.

5) Example RACE Output for Advanced Engine

Using the previous example problem information with the RACE model results with output shown in Appendix B.

F. TASK II RESULTS

As shown in the last section, the benefit and cost model, RACE, describes PPS capabilities and suitability to a specific mission model. A cost algorithm based on parametrics was developed for three major subroutines - primary propulsion costs, launch costs, and deployment operations cost. The

benefit algorithm development was based on weighted criteria ratings. The resulting benefit/cost model (RACE) is flexible and user oriented. The program models costs and benefits for STS launch and orbit transfer of any mission catalog.

Costs to execute the model is approximately 0.07¢ per thrust level iteration. The model has been verified and validated.

TABLE III-5 - ADVANCED PPS BENEFIT CRITERIA WEIGHTING AND RATING VALUES

CRITERIA	WEIGHTING	RATING
Mission Capture	65	0.0
Engine Reliability	10	9.3
Technical Risk	5	3.0
Growth Potential	5	4.0
Length of Development	5	5.0
Technical Desirability	0	5.0
Stage Reliability	10	5.0
Stage Length	0	0.0
Fabricability	0	4.0
Repairability (In Orbit)	0	0.0

TABLE III-6 - ADVANCED PPS MASSES

Propulsion Subsystem	mass, lbm	kg
Telemetry, Tracking and Command	352.0	160.0
Attitude Control	440.0	200.0
Electrical Power Supply	792000.0 lb-watts	360,000.0 kg-watts
Propellant Tank		
LH_2	374.0	170.0
LO_2	198.0	90.0
Structure	111.0	505.0
Feed and Dump System	429.0	195.0
Pressurization System	459.0	209.0
Passive Thermal Control	567.0	258.0

IV. TASK III - SAMPLE PROBLEM SOLUTION USING BENEFITS VERSUS COST ANALYSIS TECHNIQUE

A. RACE DOCUMENTATION

A complete set of user's instructions for the RACE model is presented in Appendix B. User's instructions consist of input format and description, program listing, and variable definition.

B. SAMPLE PROBLEM INPUTS FOR PRIMARY PROPULSION SYSTEMS

A sample problem was evaluated using RACE to evaluate three PPS. The inputs for this sample problem are described in the following paragraphs.

1) Propulsion Subsystem Masses

The cost of each propulsion subsystem is a function of the subsystem mass (kg) except for the electrical power supply which depends upon the product of the power supply mass and wattage. Table IV-1 shows the subsystem masses assigned for each of the engine candidates. These masses were used to calculate the RDT&E stage (without engine) cost and first unit (without engine) cost. It was recognized that some of these subsystems such as propellant tank, structure, feed and dump, and passive thermal control vary with thrust level, however an average mass across the engine thrust range was assumed for this study.

2) Weighting Factors of Benefit Criteria

The ten benefit criteria previously mentioned in Section III are shown in Table IV-2 with the weighting assigned for the sample problem. These weighting factors were supplied by the NASA contract manager. The highest weighting was assigned to mission capture (65%). In addition weightings of 10% were assigned to engine reliability and stage reliability. The other three parameters

Table IV-1 - Propulsion Subsystem Masses

Engine Candidate / Subsystem	Uprated RL-10	Advanced Engine	Dedicated Low Thrust Engine
Telemetry, Tracking and Control-kg (lbm)	160 (350)	160 (350)	160 (350)
ACS Components kg (lbm)	200 (440)	200 (440)	200 (440)
Electrical Power Supply-kg x Watts (lbm x Watts)	180 kg x 2000 W 360,000 (792,000)	360,000 (792,000)	360,000 (792,000)
LH_2 Tank-kg (lbm)	169 (372)	170 (374)	172 (376)
LO_2 Tank-kg (lbm)	90 (198)	90 (198)	92 (202)
Structure-kg (lbm)	505 (1110)	505 (1110)	505 (1110)
Feed and Dump-kg (lbm)	195 (430)	195 (430)	195 (430)
Pressurization-kg (lbm)	245 (540)	209 (460)	209 (460)
Passive Thermal Control-kg (lbm)	256 (563)	258 (568)	260 (572)

which bring the weighting factor total to 100% are technical risk, growth potential, and length of development phase. Benefit criteria which did not receive any weighting are technical desirability, stage length, fabricability, and repairability (in orbit).

Table IV-2 - Benefit Criteria Weighting Factors
For Sample Problem

Criteria	Weighting Factor
Mission Capture	65
Engine Reliability	10
Technical Risk	5
Growth Potential	5
Length of Development	5
Technical Desirability	0
Stage Reliability	10
Stage Length	0
Fabricability	0
Repairability (in orbit)	0

3) <u>Benefit Criteria Rating of each PPS</u>

To complete the input for the model it was necessary to assign benefit criteria rating for each PPS for the benefit section of RACE. The ratings are unique for each PPS and are based on a 0 to 10 scale where a rating of 10 is best. The benefit criteria, a numeric rating and its corresponding verbal definition are presented for each PPS.

a) <u>Uprated RL-10 Benefit Ratings</u>

Mission Capture:	Calculated by RACE.
Engine Reliability:	9.0 - 0.997 Reliability (NASA supplied).
Engine Technical Risk:	6.0 - slightly less than major modification to existing design.
Engine Growth Potential:	1 - almost total redesign required for significant performance increase since engine has been modified three times already.

Engine Length of Development:	8 - 4 years development period.
Engine Technical Desirability:	8 - improved performance from existing technology.
Stage Reliability:	5 - 0.980 (based on stage reliability of the advanced spacecraft propulsion design, Contract No. F04611-81-C0046).
Stage Length:	Calculated by RACE per rating scale.
System Fabricability:	5 - major modification to existing manufacturing techniques for torus tank.
Repairability (In Orbit):	0 - system is not repairable in orbit due to safety considerations.

b) Advanced Engine Benefit Ratings

Mission Capture:	Calculated by RACE.
Engine Reliability:	9.3 - 0.998 reliability (NASA supplied).
Engine Technical Risk:	3 - new design, state of the art.
Engine Growth Potential:	4 - slightly less than a major modification since higher thrust level has more flexibility to modify.
Engine Length of Development:	5 - 7 years development period.
Engine Technical Desirability:	5 - new use of existing technology.
Stage Reliability:	5 - 0.980 (based on stage reliability of the Advanced Spacecraft Propulsion Design, Contract No. F04611-81-C0046).
Stage Length:	Calculated by RACE per rating scale.
System Fabricability:	4 - more than major modification to existing manufacturing techniques for torus tank and new engine design.
Repairability (In Orbit):	0 - system is not repairable in orbit due to safety considerations.

c) Dedicated Low Thrust Engine Benefit Ratings

Mission Capture:	Calculated by RACE.
Engine Reliability:	9.3 - 0.998 (NASA supplied).
Engine Technical Risk:	2 - state of the art plus complex turbo machinery.
Engine Growth Potential:	3 - slightly more than a major modification since low thrust level has less flexibility to modify.
Engine Length of Development:	4 - 8 years development period.
Engine Technical Desirability:	7 - new use of existing technology with breakthrough in turbo machinery.
Stage Reliability:	5 - 0.980 (based on stage reliability of Advanced Spacecraft Propulsion Design, Contract No. F04611-81-C0046).
Stage Length:	Calculated by RACE per rating scale.
System Fabricability:	4 - more than major modification to existing manufacturing techniques for torus tank and new engine design.
Repairability (In Orbit):	0 - system is not repairable in orbit due to safety considerations.

4) Liquid Rocket Engine Costs

The final input information necessary to exercise the RACE model is each liquid rocket engine RDT&E cost and first unit cost. These values which are shown in Table IV-3 were furnished by the NASA contract manager.

Table IV-3 - Liquid Rocket Engine Costs (1982 Dollars)

Cost / Engine	RDT&E $(Millions)	First Unit $(Millions)
Uprated RL-10	105	2.0
Advanced Engine	270	2.8
Dedicated Low Thrust Engine	253	2.4

5) Mission Model Information

One of the Task I results was a LSS mission model which can be compared to each engine/stage to define the mission capture. This mission model information included payload weight, acceleration range, and number of stages required to deliver the payload to GEO. Additional mission model inputs included payload weight, acceleration range, and number of stages required to deliver the payload to GEO. Other inputs required by RACE are mission probability and application type. Mission probability is defined as the likelihood of mission occurrence. Application type refers to whether the payload is commercial or government oriented. Table IV-4 presents the mission probability and application for both the NASA/Commercial and DOD mission catalogs.

C. SAMPLE PROBLEM OUTPUT FORMAT

When using the RACE model, the output can be expressed in various formats, each revealing valuable insight of the three PPS comparisons. The four formats which describe the PPS benefits/cost and their comparison are: 1) PPS mission capture rating versus thrust level; 2) PPS benefit versus thrust level; 3) LCC/stage/benefit versus thrust level; 4) LCC versus percentage of stages captured.

PPS parameters of the first two formats are normalized such that values range from 0 to 10. Higher values of PPS mission capture rating and PPS benefit are preferred. Discontinuities of the data occur when a mission is captured or lost. As thrust increases, discontinuities that increase mission capture rating or benefit represent a mission capture occurrence. Conversely, as thrust increases, discontinuities that decrease mission capture rating or benefit represent a mission loss occurrence.

Table IV-4 - Probabilities and Applications of NASA/Commercial and DOD Mission Catalog

Mission	Probability	Application
1) Electronic Mail Transmission-Demonstration	1.0	Government
2) Near-Term Navigation Concept	0.95	Government
3) Demonstration Geosynchronous Platform	1.0	Commercial
4) Electronic Mail Transmission	1.0	Government
5) Technology Development Platform	0.5	Government
6) Full-Capacity GSO Communication Platform	1.0	Commercial
7) Voting/Polling Wrist Set	0.2	Government
8) Energy Monitor	0.1	Commercial
9) Orbital Antenna Farm	0.85	Commercial
10) Personal Navigation-Wrist Set	0.2	Commercial
11) Marine Broadcast Radar	1.0	Government
12) Orbiting Deep Space Relay Station	0.7	Government
13) Personal Communication-Demonstration (Wrist Set)	0.8	Commercial
14) Disaster Communications Satellite	0.5	Government
15) Police Communications Satellite	0.3	Government
16) Burglar Alarm Relay Satellite	0.5	Commercial
17) Space Based Radar-Far Term	1.0	Government
18) Security Surveillance of Unmanned Sites	0.3	Government
19) Distress Signal Pinpointing	0.3	Government
20) Classified	1.0	Government
21) Classified	0.9	Government
22) Classified	0.6	Government
23) Classified	0.1	Government
24) Classified	1.0	Government
25) Classified	0.95	Government
26) Classified	0.5	Government
27) Classified	0.9	Government
28) Classified	0.8	Government

The third data format is LCC/stage/benefit versus thrust. The most favorable PPS will have the maximum number of captured stages, maximum benefit, and minimum LCC. Thus the lower the LCC/stage/benefit value the better.

The first three output formats do not directly address LCC. The fourth format, LCC versus percentage of stages captured, will present the best capture percentage of each PPS and the corresponding LCC.

D. SAMPLE PROBLEM RESULTS

A benefits/costs comparison of the three PPS was conducted for three mission catalogs. The results are presented for the NASA/Commercial, DOD, and NASA/Commercial/DOD respectively. These results are dependent on the mission catalog. Other mission catalog inputs will result in different conclusions.

1) NASA/Commercial Catalog Results

A benefits and costs comparison of the three PPS resulted when the 16 mission NASA/Commercial catalog was coupled with the appropriate RACE input data. The results are presented in Figures IV-1 through IV-4.

Mission capture rating versus thrust is shown in Figure IV-1. The general decreasing of the mission capture rating as thrust level increases agrees with the requirement that mission capture probability decreases as thrust level increases. A rating of 10 implies all the missions are captured at or below the most conservative acceleration level. A rating of 5 can correspond to half of the missions captured at or below the most conservative acceleration level or all the missions capture at a probability of 50%. Actually a mixture of mission acceleration ranges are reflected in the PPS benefit values. Sixty-eight PPS are required to capture the 16 mission NASA/Commercial catalog. Mission capture rating greatly decreases above 30,000N for the advanced engine and the uprated RL-10 engine.

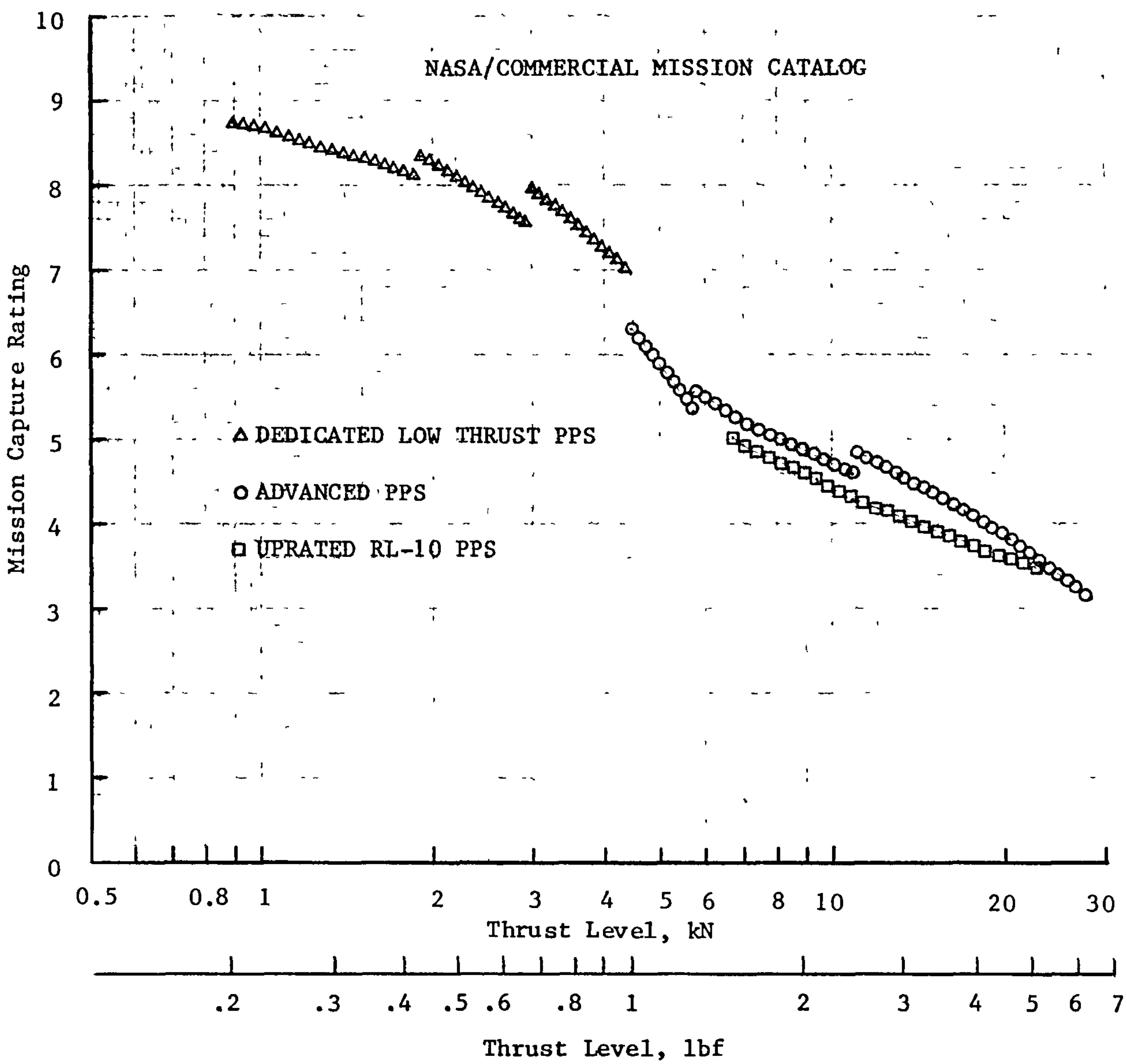


FIGURE IV-1 MISSION CAPTURE RATING VERSUS THRUST LEVEL (NASA/COMMERCIAL)

The Dedicated Low Thrust Engine captures the maximum number of stages, 61 of 68, between thrust levels of 3000 to 4450 N. An Advanced Engine can capture 54 stages between 4450 to 7200 N. The Uprated RL-10 can capture only 30 stages out of 68 stages. Maximum mission capture of the RL-10 occurs at either 6670 to 7280 N or 8600 to 9500 N. Ranges of mission capture rating for the Uprated RL-10, Advanced Engine and Dedicated Low Thrust Engine are 5.0 to 3.5 (6670 to 22,850N), 6.3 to 3.2 (4450 to 27,550N), and 8.7 to 7.0 (900 to 4,320N), respectively.

Similar to the mission capture rating trend, benefit rating versus thrust level appears in Figure IV-2. However, the ranges of benefit rating are compressed since mission capture is weighted 65% of the total benefit rating value. Ranges of benefit rating for the Uprated RL-10, Advanced Engine and Dedicated Low Thrust Engine are 5.4 to 3.8 (6650 to 40,000N), 6.1 to 3.8 (4450 to 40,000N), and 7.6 to 6.4 (900 to 4450N), respectively.

It was desired to develop a parameter which would accurately reflect not only the benefits but also the cost of each candidate PPS in conjunction with a specific mission model. This parameter should either be maximized or minimized for the candidate PPS that conforms best to a mission model. It was determined that the parameter which best describes the benefits and cost of an engine is life cycle cost per stage per benefit rating point. A minimum value of this parameter is desired and can be accomplished three ways. First, the LCC of a PPS can be low; second the number of stages captured for the LCC value can be large; or finally the benefit rating can be large. This parameter is graphically illustrated as a function of thrust in Figure IV-3. The minimum values for the Uprated RL-10 PPS, Advanced PPS, and Dedicated Low Thrust PPS are 13.4×10^6, 11.0×10^6, and 8.7×10^6, respectively.

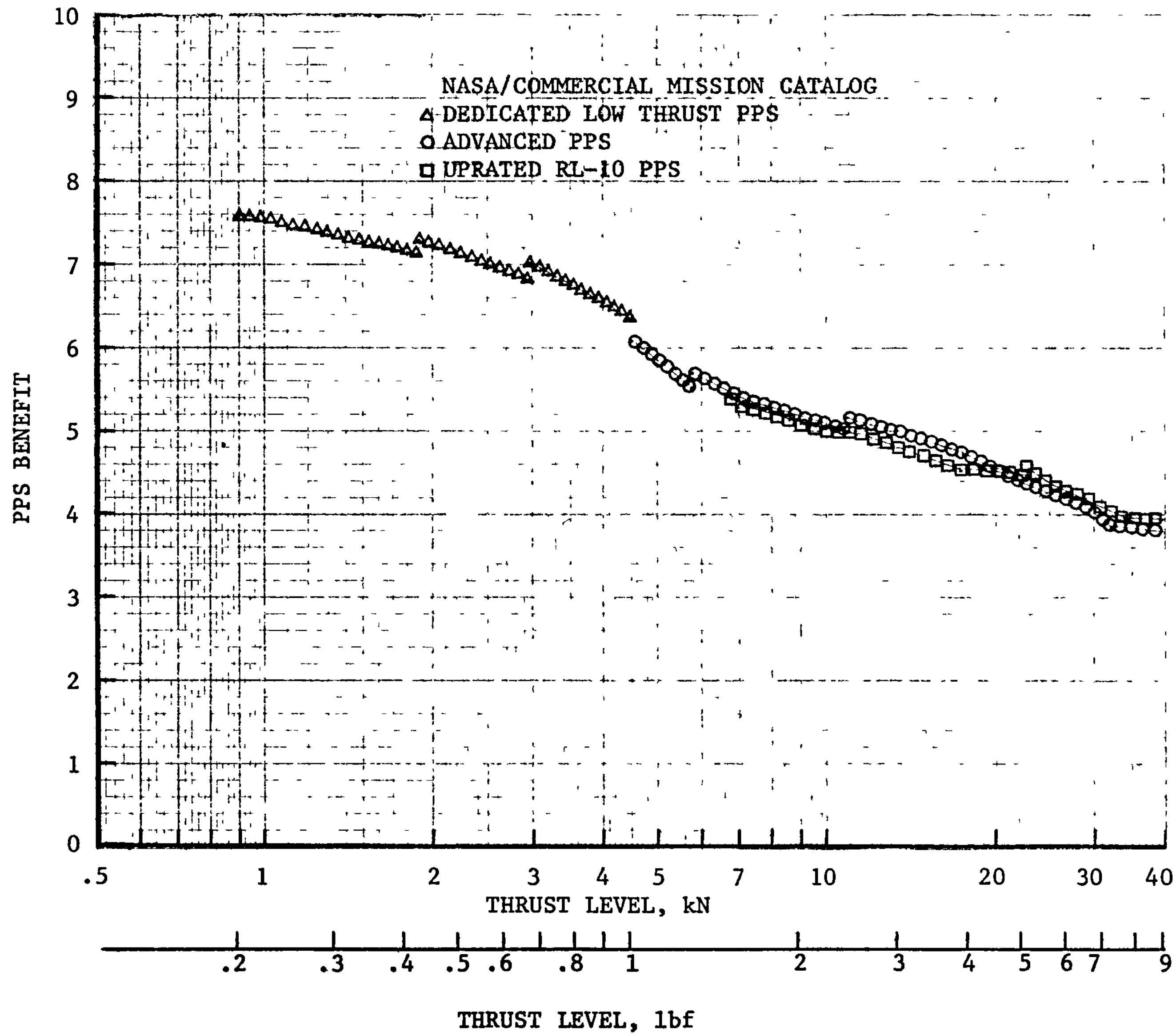


FIGURE IV-2 BENEFIT RATING VERSUS THRUST LEVEL (NASA/COMMERCIAL)

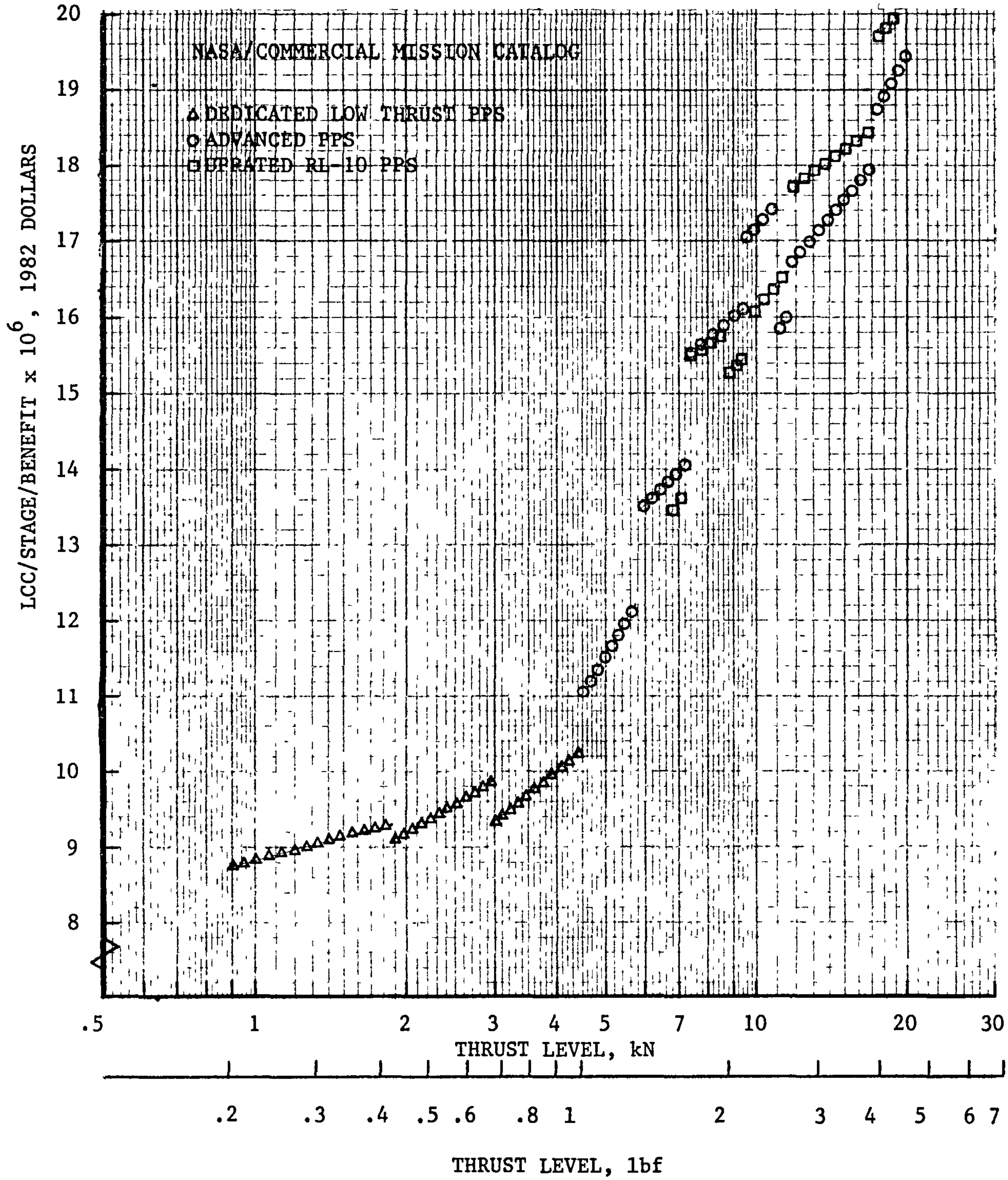


FIGURE IV-3 LCC/STAGE/BENEFIT VERSUS THRUST LEVEL (NASA/COMMERCIAL)

The three candidate propulsion systems are compared on the bases of cost and benefit in Figure IV-3. The dedicated low thrust PPS has a range of LCC/stage/benefit from \$8.7 x 10^6 to \$10.3 x 10^6. These values are the lowest of all three PPS. The second most favorable PPS uses the advanced engine. At its lowest thrust, 4450 N, LCC/stage/benefit is \$11.0 x 10^6. This parameter quickly increases to \$12.1 x 10^6 at 5700 N. The most favorable LCC/stage/benefit value (4450 N) for the advanced PPS is 25% higher than the dedicated low thrust PPS counterpart value at 890 N. The uprated RL-10 PPS has the least attractive LCC/stage/benefit value of the three candidates. Its best performances are at 6670 N to 7250 N with corresponding LCC/stage/benefit of \$13.4 X 10^6 to \$13.7 X 10^6, respectively. The most cost effective and beneficial system of the three is the dedicated low thrust PPS. Variations in LCC/stage/benefit are from \$8.7 x 10^6 to \$10.3 x 10^6. These LCC/stage/benefit values represent a 20% span across the engine thrust range. The best specific operating thrust for the dedicated low thrust PPS would be about 3000 N. Value of LCC/stage/benefit at 3000 N is \$9.3 x 10^6, which is only 6% higher than the most favorable LCC/stage/benefit value at 890 N, but the thrust of 3000 N is more than three times as great as the most favorable value.

Figures IV-1 through IV-3 has shown the results when the three candidate PPS are compared to the 16 mission non-DOD model. There are 68 stages (including replacements) in this model. Not one of the candidate PPS can capture 100% of this mission model as it is presently defined. The missions which cannot be captured either have acceleration requirements that cannot be provided by the PPS or exceed payload capability of the PPS. However, an LCC, which would reflect the cost of launching all 16 missions, could be estimated if the missions that were not captured could be relocated by some means to fall within the mission capture envelopes. Such a relocation was accomplished by considering how the spacecraft characteristics could be changed so that the mission falls within the PPS envelopes.

There were two principle reasons that the missions were not captured. Either the mass of the spacecraft was too great to be handled by the PPS or the acceleration that the spacecraft was capable of withstanding was below the lowest acceleration that the PPS can provide. This lead to two scenarios that could be postulated that would allow the spacecraft characteristics to be changed so that the mission would fall within the PPS performance envelope.

In the case of spacecrafts that were too heavy, it would be possible to bring them into the envelope by reducing their mass. Insufficient information is available to determine how the structure could be lightened and the cost of such a change. Therefore the approach was taken that heavier spacecraft would be divided into two equal mass spacecrafts. This would halve the mass of the spacecraft but would double the number of STS and PPS needed for that particular mission. The cost of the additional STS and PPS provided a cost factor that was consistent with the costing procedures used elsewhere in the program.

The following scenario was defined for those missions that were not captured because their acceptable acceleration limits were too low. It was assumed that the acceleration a spacecraft could tolerate could be increased if the structure was strengthened by the addition of material. Information from two previous studies, Primary Propulsion/Large Space System Interaction Study and Study for Auxiliary Propulsion Requirements for Large Space Systems, was used to determine the affects of increasing the acceleration limits. A ratio of one to one for percentage acceleration increase to mass increase was used to determine the effects on the spacecraft. No cost penalty was imposed on payloads if increase of mass required to strengthen the payload did not cause the payload to become too heavy for the PPS system. This was due to the ground rule that the payloads are assumed to be placed in orbit by dedicated SST's and were volume limited, i.e. the cost of the flight was already paid and no charge would be assessed for additional weight.

Table IV-5 summarizes the mission catalog information for the relocated missions. Eight missions were relocated for the Advanced Engine PPS capture. The affected missions are 2, 4, 9, 10, 11, 12, 14 and 15. Of these, missions 2, 4, 9 and 11 were successfully moved by increasing LSS strength. The remaining missions, 10, 12, 14 and 15, necessitated a payload division. Eleven mission relocations were required if all missions were to reside within the Uprated RL-10 PPS payload mass/acceleration envelopes. A total of six missions, 2, 4, 5, 7, 9 and 11 were accommodated by increased LSS strength. Missions that dictate a payload split were 10, 12, 14, 15, and 16. Few missions require displacement for the Dedicated Low Thrust PPS. Two missions, 2 and 11, were moved by strengthening the LSS. As with the other PPS, missions 14 and 15 were divided.

Moving missions by strengthening the LSS had very little cost impact. However, doubling the numbering of STS launches and PPS was a substantial cost addition considering each STS launch alone was at least $37 mission dollars. Clarification of the additional capture cost is displayed in Figure IV-4. Each PPS is represented and the cost accrued to capture each mission is illustrated. Mission 10 dominates the addtional costs for the Advanced PPS and the Uprated RL-10 PPS. Mission 10 is a multichannel switching satellite that allows two-way voice telecommunications using small Earth based wrist sets. The payload is a cross structure with each arm being 1700m x 5m. Increasing the acceleration range required the payload to be divided into 5 sections for the advanced low thrust PPS and 6 sections for the uprated RL-10 PPS. Thus a drastic increase in the number of STS launches result.

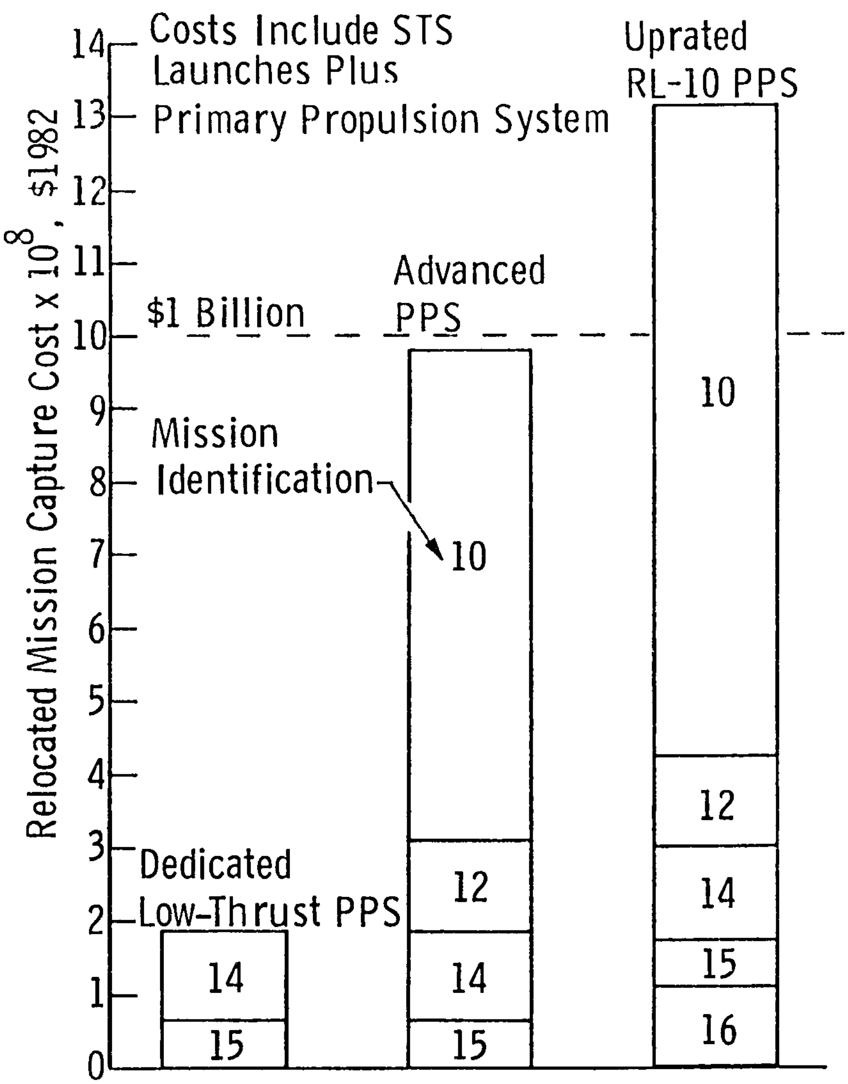


FIGURE IV-4 Relocated Mission Capture Cost

Based on the modified mission model the RACE program was used to evaluate LCC versus percentage of stages captured. Figure IV-5 shows the projected cost associated with capturing 100 percent of the mission catalog. Included in LCC is RDT&E, Production, Launch, and Deployment costs. The most expensive PPS was the Uprated RL-10 engine at a cost of $5.8 billion for 100% capture. The Advanced Engine PPS would cost $5.6 billion for 100 percent capture. Total capture for the Dedicated Low Thrust PPS would cost $4.6 billion which is the least expensive. Also shown in Figure IV-5 are the PPS capabilities for the original NASA/Commercial mission catalog. The Uprated RL-10 captures 44% of the mission catalog at a cost of $2.1 billion. The

second best capture percentage (79%) belongs to the Advanced PPS for a cost of $3.6 billion. Best capture is 90% for the Dedicated Low Thrust PPS and costs $3.9 billion.

TABLE IV-5 MISSION RELOCATION INFORMATION

ADVANCED ENGINE PPS

MISSION #	PAYLOAD MASS kg	ACCELERATION RANGE g's	NUMBER OF STS LAUNCHES
2	1680	0.13 to 0.16	1
4	4780	0.07 to 0.15	8
9	3330	0.1 to 0.25	2
10	6800	0.05 to 0.14	20
11	5270	0.09 to 0.18	16
12	3750	0.07 to 0.22	8
14	4100	0.07 to 0.37	8
15	4100	0.07 to 0.37	4

UPRATED RL-10 PPS

MISSION #	PAYLOAD MASS kg	ACCELERATION RANGE g's	NUMBER OF STS LAUNCHES
2	1850	0.16 to 0.19	1
4	4870	0.09 to 0.17	8
5	3250	0.12 to 0.22	2
7	6190	0.08 to 0.23	2
9	3350	0.13 to 0.28	2
10	6800	0.09 to 0.18	24
11	4800	0.08 to 0.11	16
12	3750	0.11 to 0.26	8
14	4100	0.12 to 0.42	8
15	4100	0.12 to 0.42	4
16	3630	0.11 to 0.41	4

DEDICATED LOW THRUST PPS

MISSION #	PAYLOAD MASS kg	ACCELERATION RANGE g's	NUMBER OF STS LAUNCHES
2	1630	0.025 to 0.055	1
11	3300	0.02 to 0.12	16
14	4100	0.05 to 0.35	8
15	4100	0.05 to 0-.35	4

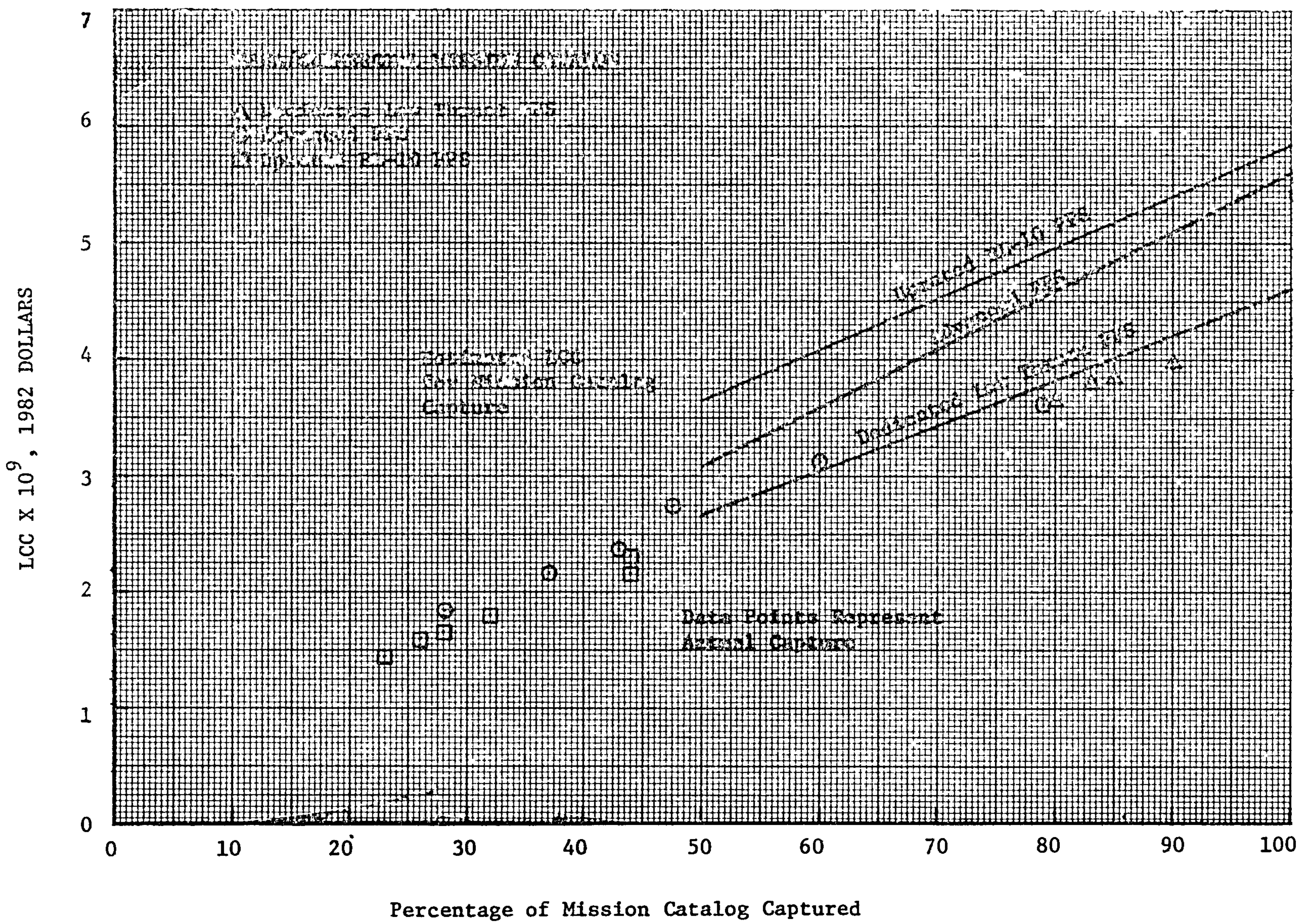


FIGURE IV-5 LCC TO CAPTURE NASA/COMMERCIAL MISSION CATALOG

2) DOD Mission Catalog Results

PPS benefit versus thrust level for DOD mission catalog is presented in Figure IV-6. Dedicated Low Thrust PPS has the highest benefit (8.3) of all candidates. Most attractive thrust level range of this PPS is between 3400 to 4450 N. Maximum benefit of the Advanced PPS is 6.5 at 7000 N. Uprated RL-10 PPS has a benefit of 4.7 at 6670 N.

LCC/stage/benefit versus thrust level for the DOD Catalog are shown in Figure IV-7. Since most DOD missions reside inside the Advanced PPS and Uprated RL-10 PPS capture envelopes, the mission capture benefit (Figure IV-6) does not begin decreasing rapidly until a thrust level of 4700 N is reached. Thus, slopes of LCC/stage/benefit for the dedicated low thrust PPS are small. Rapidly decreasing mission probabilities above 4700 N yield large slope values for the higher thrust PPS.

The most beneficial PPS is the dedicated low thrust. LCC/stage/benefit between 3380 to 4450 N is \$5.6 x 10^6 and changes very slightly. At 7000 N the advanced PPS has a LCC/stage/benefit value of 12.3 x 10^6. This LCC/stage/benefit value for the RL-10 PPS is 120% higher than the best dedicated low thrust PPS performance.

The DOD catalog results presented in Figure IV-7 support the Dedicated Low Thrust PPS as being the most beneficial and cost effective PPS of the three candidate compared. Furthermore the most attractive thrust level range for which the Dedicated Low Thrust engine should operate is between 3380 to 4450 N. The number of stages captured in this thrust range is 198 out of 202 at a LCC value of \$9.26 X 10^9.

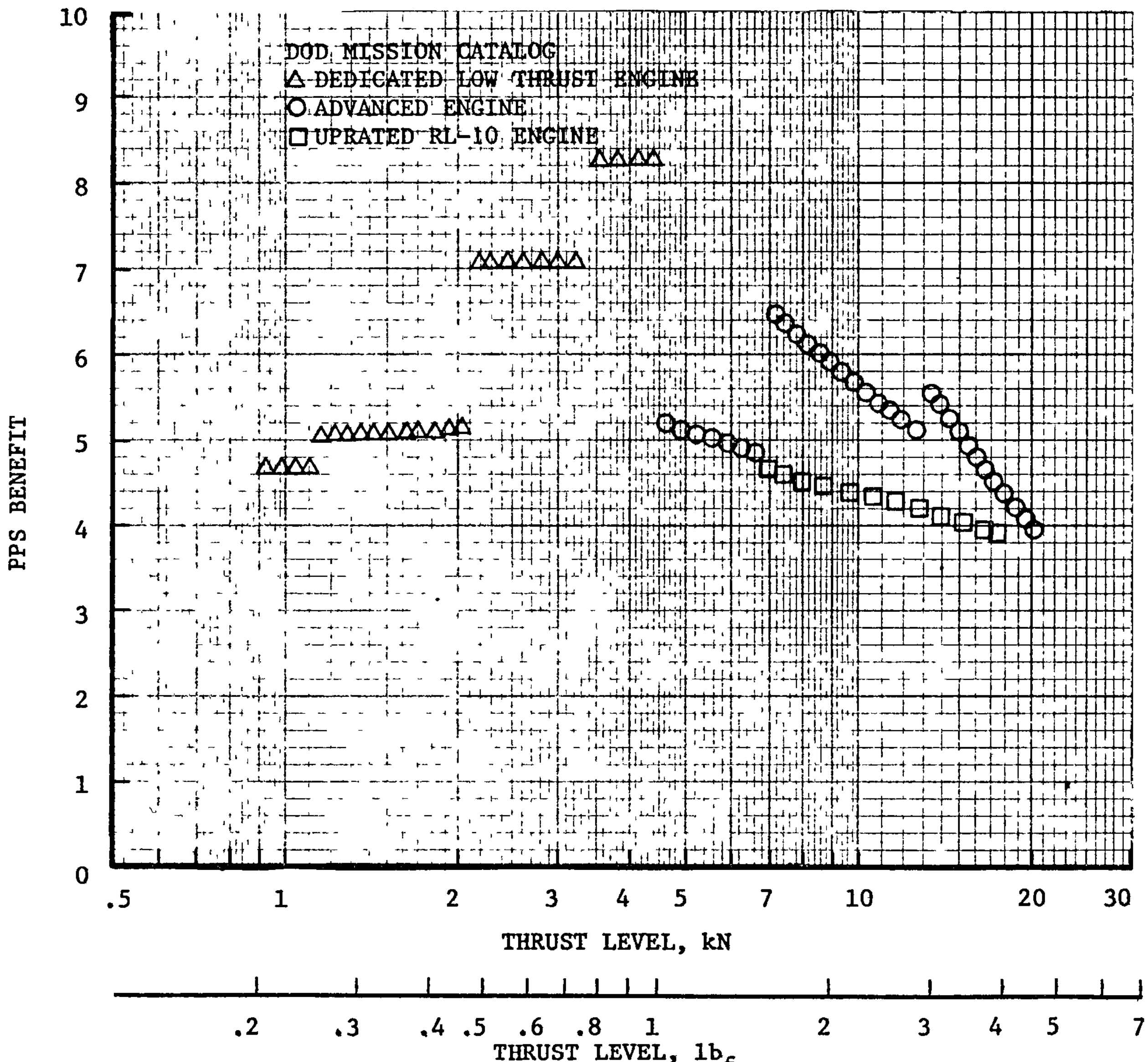


FIGURE IV-6 PPS BENEFIT VERSUS THRUST LEVEL (DOD)

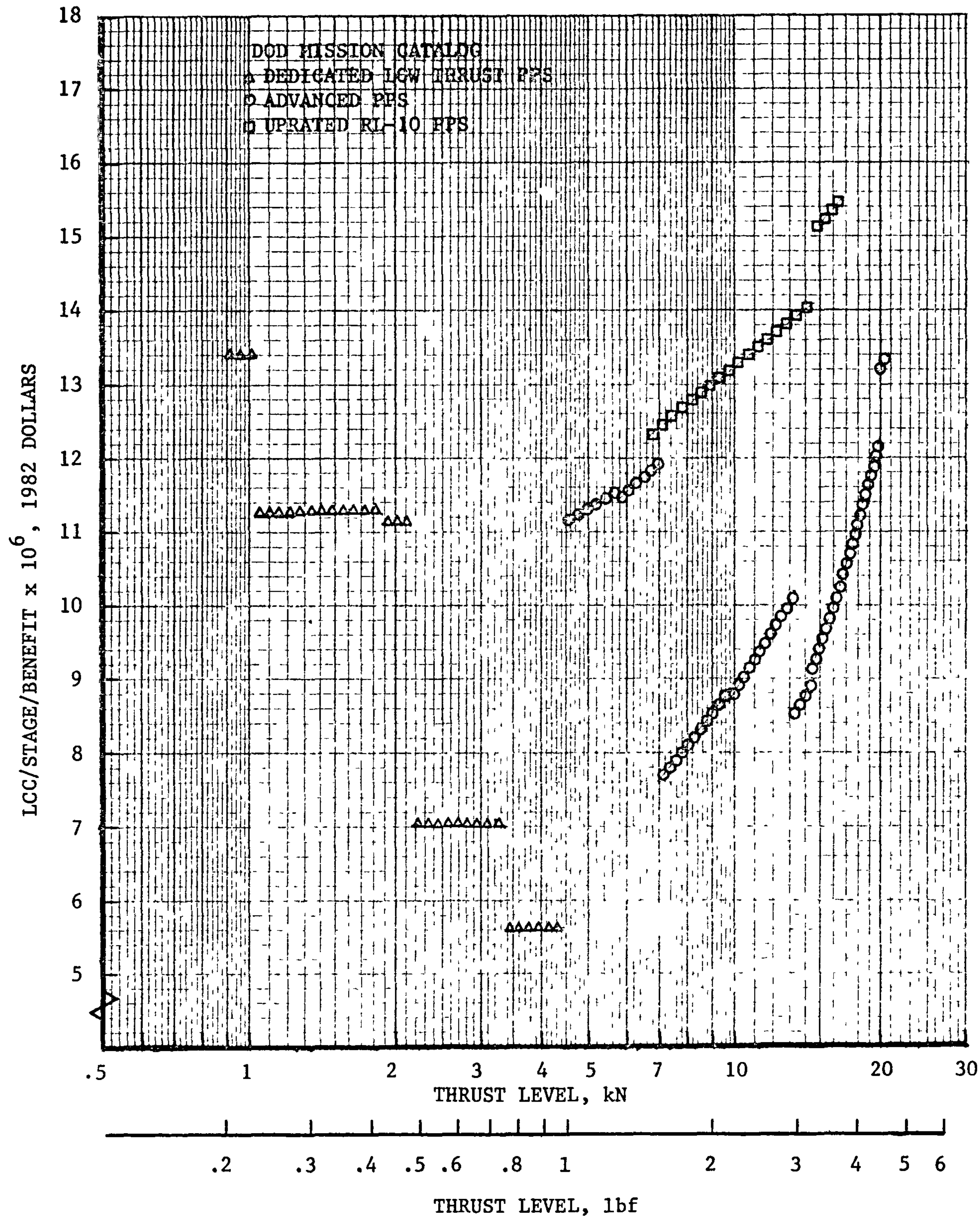


FIGURE IV-7 LCC/STAGE/BENEFIT VERSUS THRUST LEVEL (DOD)

3) Combined DOD and NASA/Commercial Mission Catalog Results

RACE results of the two LSS catalogs combined are shown in Figure IV-8. As expected the Dedicated Low Thrust PPS performed the best of all. Best values of LCC/stage/benefit are $6.0 x 10^6 at 3380 N to $6.2 x 10^6 at 4450 N. Results for the combined catalogs resemble DOD catalog results since the DOD catalog requires four times as many PPS as the NASA/Commercial model.

These results support the development of a dedicated low thrust PPS with a thrust level between 3380 N to 4450 N. This PPS is the most cost effective and beneficial for these LSS mission catalogs.

4) Effects of Advanced Lightweight Structures

One task of program was to determine the effects of advanced lightweight structures on the candidate PPS parameter, LCC/stage/benefit point. Improvements in lightweight structures will mainly rely upon material technology breakthroughs in the form of improved strength. Materials with increased strength will allow LSS to be lighter or to be transferred from LEO to GEO at higher accelerations (higher thrust). A 20% structural strength increase was selected to determine the impact on LCC, engine selection, and thrust level. It is believed that this 20% increase represents the maximum realistic improvement in this timeframe. Only the NASA/Commercial mission catalog was used for the comparison.

The approach used was to either increase the acceptable acceleration for each mission or reduce the structure mass of the spacecraft. From previous work it was known that an increase in strength is approximately proportional to an increase in acceleration given the same size structure - this assumption will result in no more than 10% error. Using this improvement, end points of the acceleration ranges for each LSS were increased by 20% if the spacecraft mass falls below the maximum payload mass/acceleration capabilities of the engines. However, missions that fall outside the

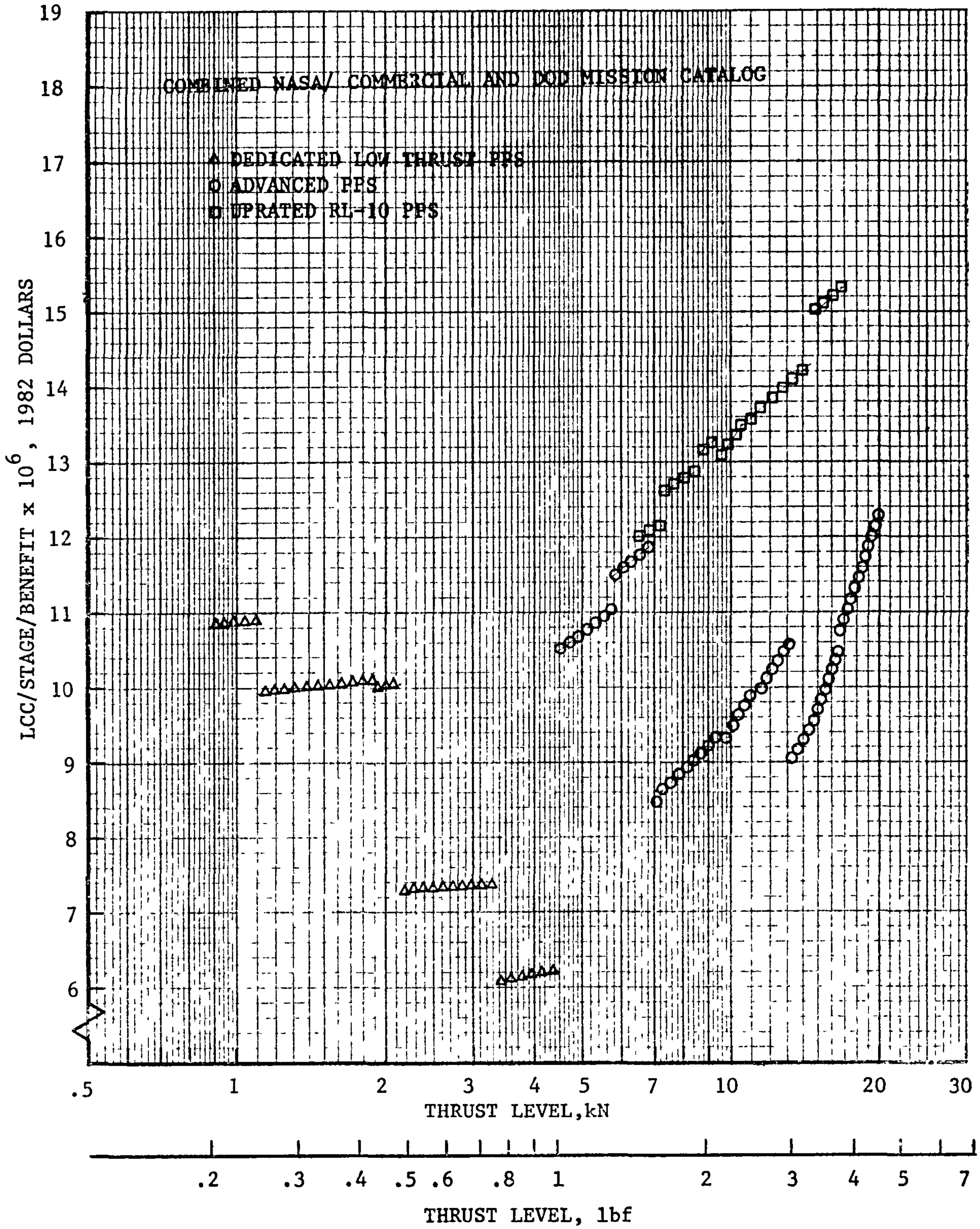


FIGURE IV-8 LCC/STAGE/BENEFIT VERSUS THRUST LEVEL FOR NASA/COMMERCIAL AND DOD MISSION CATALOG

envelopes (12, 14 and 15) had their structural mass reduced by 20% to improve the probability of capture.

The impact of advanced lightweight structures is shown in Figure IV-9, LCC/stage/benefit versus thrust level. All three candidate PPS are represented in Figure IV-9 and results are shown for current structural strength (solid symbol) and the 20% strength increase (open symbol). As expected, the adjusted mission model, which incorporates advanced structure, has a lower value for LCC/stage/benefit than the original mission model since higher allowable accelerations lead to a higher mission capture rating which in turn results in a higher benefit value.

There are two views of the results. First, for a constant value of LCC/stage/benefit then the advanced structures missions can be transferredfrom LEO to GEO at a thrust level up to 300% greater than the original missions. The percentage increase to thrust varies with the engine thrust level, with the greatest increasesoccurring in the flat region of the data (1000 to 3500 N thrust). The second conclusion that can be drawn from the results is the cost benefit of advanced structures. Depending upon the thrust level of candidate PPS selected, one-half to two million dollars per stage could be saved with the advanced materials. Since the NASA/Commercial mission model consists of a total of 68 stages, advanced material could reduce the overall program cost by 134 to 135 million dollars depending upon selected thrust level.

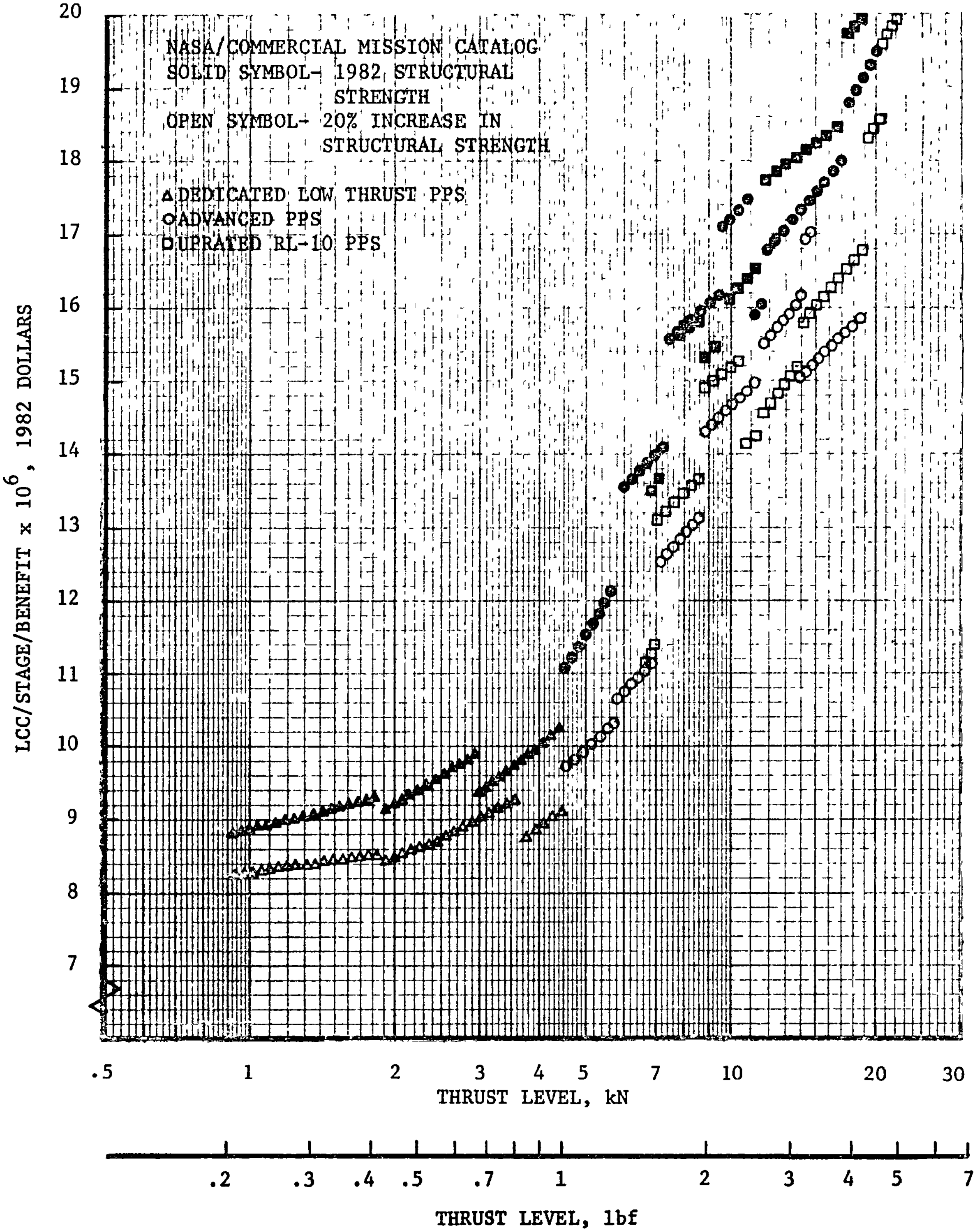


FIGURE IV-9 EFFECTS OF ADVANCED MATERIALS ON LCC/STAGE/BENEFIT

V. RESULTS

This program investigated the benefit and costs associated with placing LSS in operational orbits and developed a flexible computer model for analyzing these benefits and costs. The primary tasks of the program were to define the LSS mission characteristics, develop a cost benefit analytical model and solve a sample problem utilizing the technique developed.

A. LSS MISSION CHARACTERISTICS

A mission model for LSS from the present time thru the year 2010 was established. Current mission models and current literature were reviewed to determine the LSS missions that are forseen for the years in question. The types of structures that would be used by the LSS for each mission were also identified. Initially, only the NASA/commercial missions were included in the model. However, an additional work effort was added to the program to include the DOD missions. A total of sixteen NASA/commercial missions and twelve DOD missions were identified as falling within the ground rules of the study to be included in the mission model. The number of STS launches required to accomplish any one mission varied from one to twenty-five. This creates a requirement for a total of 68 STS to launch the NASA/commercial missions and 202 STS to launch the DOD missions.

The mission catalog was defined in sufficient depth to allow the mass of each payload and the acceleration limits that it could tolerate to be defined. The masses of the LSS ranged from 1600 kg to 8200 kg. The acceptable final acceleration limits varied from 0.01 to 1.0 g.

Conceptual primary propulsion stage designs were developed for the three low thrust engines that were baselined for the study, a Dedicated Low Thrust engine with thrust capablity from 890-4450 N(200-1000lbf), an Advanced engine with thrust capability of 4450-66700 N (1000-15,000lbf) and an Uprated RL-10 engine with thrust capablity of 6670-15000 N (1500-15,000 lbm). The stage designs were based on a minimum length

configuration using a toroidal tank with an embedded engine to allow the maximum volume of the orbiter payload to be used for the LSS spacecraft. The transfer vehicle was then sized for various initial masses of the mated PPS/LSS to develop mission capture envelopes for each engine.

The mission catalog was then compared to the PPS performance to create a mission capture. This mission capture showed that none of the PPS could capture all of the missions. The PPSs using the Dedicated Low Thrust engine and the Uprated RL-10 do not have performance high enough to capture some of the heavier LSS, while minimum accelerations of the PPS using the Advanced engine and the Uprated RL-10 were too high for some of the weaker LSS. It was also determined that, for any engine, no single thrust level would capture all of the missions. To capture the maximum number of missions it would be necessary for the engines to have variable thrust capability.

B. COST/BENEFIT ANALYSIS MODEL

The costs involved in placing the LSS in their operational orbits were identified. There were two primary areas of cost. The first was the cost of the primary propulsion system including RDT&E and the second was the launch costs including the cost of the shuttle and the operational costs involved while the payload is being placed in orbit. No attempt was made to calculate the cost of the LSS. This was considered to be a fixed cost and is not affected by the type of engine that is used to place it in orbit.

When the costs had been identified, the basic cost relationships and the algorithms that could be used for describing them were established. Primarily these were broken down into three major relationships: the stage costs that are a function of design parameter; launch costs that are a function of the number of launches, and the deployment operation costs that are a function of the time required to deploy the payloads to orbit.

The second effort in developing the model was to define the benefit criteria for the mission model. The primary benefit criteria that was defined was mission capture. The more missions captured by a candidate PPS the better the PPS. This is actually a cost avoidance factor since the cost of modifying the payload, the cost of developing a new PPS that can capture the additional missions or the cost of additional STS launches are avoided. It was also recognized that other parameters can be used to evaluate the benefits of a PPS. Therefore provisions were made to include these in the model. The benefits parameters related to the engine that were included in the model were reliability, growth potential, technical risk, development time, and technical desirability. Parameters related to the stage that were included were reliability, length, repairability and fabricability.

Definitive guidelines were established for rating each of the parameters to increase the objectivity of the model. Then to give the program more flexibility, each parameter was assigned a weighting factor. The weighting factor can be adjusted to reflect the importance attached to any given evaluations. For instance, one study may need to examine the effect if reliability is the more important than mission capture. In that case the weighting factor for reliability can be larger than the mission capture weighting factor. Rating factors are function of the PPS being evaluated while weighting factors are the same for each PPS being compared and are set by the requirements of the program.

The benefit and cost relationships were then programmed into a computer model that determines the ratings and costs of each engine The inputs to the program include the PPS type, engine thrust range, mission information, benefit criteria and weighting factors and the PPS subsystem masses. The program evaluates this information on the basis of the rating and costing criteria and outputs, RDT&E costs, production costs, launch costs, LCC, number of PPS required, mission capture, and benefit values; all as a function of thrust level.

The program was verified and validated to insure that the algorithms and relationships were properly included in the model and that the model was operating properly. A copy of the program was supplied to NASA LeRC and has been running successfully on their computer.

C. SAMPLE PROBLEM SOLUTION

A sample problem was evaluated using the Benefit/Cost computer model (RACE). This sample problem was a comparison of the engines that were baselined for this study. The values for the criteria ratings and weighting factors were agreed upon between NASA LeRC and Martin Marietta. Engine cost figures were supplied by the NASA LeRC. The results of this evaluation showed that, for the conditions specified, the dedicated low thrust engine had the most favorable benefit versus cost rating.

The best indication of this was the comparison of cost per stage per benefit rating factor versus thrust level for the different PPS. It was necessary to use this complex factor for the evaluation in order to obtain a realistic rating for each engine. A comparison of the PPS benefit number at various thrust levels give an indication of the rating of each engine but the dollar figure must be included to account for cost. A comparison of cost per thrust level is misleading since there is no indication of the number of missions that would be captured. Since the cost comes down as fewer missions are flown, the most favorable thrust level as far as total cost is the one where the fewest missions are captured. Therefore, it is necessary to normalize any comparitive curve to a common base. In this case, the best normalized curve is based on life cycle cost per stage benefit rating per number of LSS captured.

To show the flexibility of the RACE program several additional problems were examined. The first problem examined was the effect that improved LSS structures would have on the results of the sample programs. It was assumed that improvements in structural materials would increase the strength of the LSS by 20%. This in effect increased the

final acceleration limits or decreased structural mass. The RACE program showed that under these conditions the LCC/benefit/stage number was reduced for each thrust level compared to the baseline program. However, it did not change the basic indication that the dedicated low thrust engine was the most attractive engine.

The RACE program was also used to examine the effect of modifying the mission model so that each of the baseline engines could capture 100 % of the mission model. The approach taken for this case was to assume that the LSS that were not captured by a given PPS could be strengthened or divided in some manner that would move their mass/acceleration characteristics within the performance envelope of the particular PPS. The results of this analysis indicated that the dedicated low thrust was still the most attractive engine with the cost of capturing 100% of the missions being far less than that of the other engines.

VI. CONCLUSIONS

This project developed a flexible computer model for evaluating the benefits and costs for launching and orbit transfer of any mission catalog. The model at present contains the performance envelopes of three primary propulsion systems for orbit transfer based on three low thrust engines baselined in the statement of work. However, it is possible to modify the basic model to examine any propulsion system. The model also allows for any mission model to be input into the program. The model presently allows the user to easily vary the program to examine the effects of various ratings and weighting of benefit parameters for the baseline engines.

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APPENDIX B - RACE PROGRAM

A. RACE Input Format

All of the information discussed in the previous section is input data for the RACE model. The model computes the benefits and costs of each engine/stage separately thus three distinct data sets are required (one for each engine). General structure of RACE input data is illustrated in Table B-1. The four major input sections are engine, LSS mission model, benefit criteria, and propulsion subsystem masses. The format of the input variables is shown in Table B-2. Above each variable field is the variable name which corresponds to its definition shown in the program listing. Below each variable field in Table B-2 is the format code that the computer recognizes for that variable. Column numbers are labeled at the top to aid the user with proper alignment of input data. The third input line identifies the LSS mission so that each mission in a mission model must be represented by a unique input line. Input of the mission identification lines can be arranged by any order. Input lines four and five represent the benefit criteria ratings and weighting factors, respectively. The last line of the data input defines the propulsion subsystem masses.

TABLE B-1 - RACE INPUT STRUCTURE

Line Numbers	Section
1 to 2	Engine
3 to N	LSS Mission Model
N+1 to N+2	Benefit Criteria
N+3	Propulsion Subsystem Masses

TABLE B-2 RACE INPUT FORMAT

```
         1         2         3         4         5         6         7
123456789012345678901234567890123456789012345678901234567890123456789012
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ENGINE TYPE	LOW THRUST LEVEL VALUE	HIGH THRUST LEVEL VALUE	THRUST LEVEL INCREMENT	NUMBER OF MISSIONS
0	00000	00000	00000	00
2XI1	2X I5	2X I5	2X I5	2X I2

	LIQUID ROCKET ENGINE RDT&E COST		LIQUID ROCKET FIRST UNIT COST
	000.00E+00		00.00E+00
10X	E10.2	10X	E9.2

MISSION NUMBER	PAYLOAD MASS	MOST CONSERVATIVE ACCELERATION	LEAST CONSERVATIVE ACCELERATION	PROBABILITY	NUMBER OF STAGES	APPLICATION
00.0	00000.0	0.000	0.000	0.00	00.0	0.0
3X F4.1	3X F7.1	3X F5.3	3X F5.3	3X F4.2	3X F4.1	3X F3.1

RATINGS

MISSION CAPTURE	ENGINE RELIABILITY	TECHNICAL RISK	GROWTH POTENTIAL	LENGTH OF DEVELOPMENT	TECHNICAL DESIRABILITY	STAGE RELIABILITY	STAGE LENGTH	FABRICABILITY	REPAIRABILITY (IN ORBIT)
000000	000000	000000	000000	000000	000000	000000	000000	000000	000000
F6.2	F6.2	F6.2	F6.2	F6.2	F6.2	F6.2	F6.2	F6.2	F6.2

WEIGHTING FACTORS

MISSION CAPTURE	ENGINE RELIABILITY	TECHNICAL RISK	GROWTH POTENTIAL	LENGTH OF DEVELOPMENT	TECHNICAL DESIRABILITY	STAGE RELIABILITY	STAGE LENGTH	FABRICABILITY	REPAIRABILITY (IN ORBIT)
000000	000000	000000	000000	000000	000000	000000	000000	000000	000000
F6.1	F6.1	F6.1	F6.1	F6.1	F6.1	F6.1	F6.1	F6.1	F6.1

MASSES

TELEMETRY TRACKING & CONTROL	ATTITUDE CONTROL	ELECTRICAL POWER SUPPLY	LH_2 PROPELLANT TANK	LO_2 PROPELLANT TANK	STRUCTURE	FEED AND DUMP	PRESSURIZATION	PASSIVE THERMAL CONTROL
000.0	000.0	000000.0	000.0	000.0	000.0	000.0	000.0	000.0
2XF5.1	2XF5.1	2X F8.1	2XF5.1	2XF5.1	2XF5.1	2XF5.1	2XF5.1	2XF5.1

B. RACE Program

Table B-3 lists the latest version of RACE. The language of RACE is FORTRAN IV. Verification of this model is complete, however its length prohibits publication. Approximate cost to execute this model once compiled is 0.07¢ per thrust level iteration. Therefore, the cost of a 6000 iteration data set with mission model of size 17 is less than $4.50. The program flow chart is shown in Table B-4.

C Example RACE Output

An example of the output format and ability that RACE will deliver is shown in Table B-5. Page 1 of Table B-5 reiterates the engine type, thrust range of investigation, mission information, and benefit criteria information. All information shown on Page 1 is input data and is printed for verification. The first item on top of Page 2, propulsion subsystem masses, completes all input data except engine cost which appears at the bottom of the page.

The first output is the Mission Model Matrix. These results are independent of the engine selection. The most conservative thrust and least conservative thrust correspond to a most conservative payload acceleration and least conservative payload acceleration, respectively. The minimum thrust to deliver a payload is the thrust level where payload capability of the engine/stage combinations drops below payload mass. The last two columns of mission model matrix refer to the psi (Ψ) function. The psi function (mission capture index) is a linear probability of the actual LSS design acceleration across the payload final acceleration range. As the final acceleration approaches the most conservative payload acceleration, psi approaches a value of 1.0.

The bottom of the second page, Table B-5, shows the engine input costs (RDT&E and First Unit) plus the RDT and E cost and First Unit cost of the stage without the liquid rocket engine. PPS cost is the sum of these respective values. All costs are reported in millions of 1982 fiscal year dollars.

TABLE B-3 RACE PROGRAM

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      PROGRAM RACE (INPUT,OUTPUT,TAPE2=INPUT,TAPE3=OUTPUT)
C       RACE- RATING AND COST OF ENGINE
C     WRITTEN UNDER CONTRACT MCR-82-500 FOR NASA LEWIS RESEARCH CENTER
C     STUDY FOR ANALYSIS OF BENEFIT VERSUS COST OF LOW THRUST PROPULSION
C     PROGRAM BENE IS WRITTEN SPECIFICALLY FOR AN UPRATED RL-10, ADVANCED
C     ENGINE,OR DEDICATED LOW THRUST ENGINE
C
C XXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXX
C
C     VARIABLES OF RACE ARE DEFINED BELOW
C      ACCEL-APPROXIMATE FINAL ACCELERATION LEVEL
C      ACS-ATTITUDE CONTROL SYSTEM MASS
C      ACSRDT-ATTITUDE CONTROL SYSTEM RDT AND E COST
C      ACSUNIT -ATTITUDE CONTROL SYSTEM UNIT COST
C      ACSWT-ATTITUDE CONTROL SYSTEM WEIGHT (LBS)
C      APROCOS-AVERAGE PRODUCTION COST
C     BENEFIT=ENGINE BENEFIT RATING AT SPECIFIC THRUST LEVEL
C     BENEVAL=A 2 X 10 MATRIX CONTAINING ENGINE BENEFIT INFORMATION
C             ROW N=1 IS BENEFIT PARAMETER RATING
C             ROW N=2 IS BENEFIT PARAMETER WEIGHTING FACTOR
C       BENEVAL(N,1)=MISSION CAPTURE
C       BENEVAL(N,2)=RELIABILITY
C       BENEVAL(N 3)=TECHNICAL RISK
C       BENEVAL(N,4)=GROWTH POTENTIAL
C       BENEVAL(N,5)=LENGTH OF DEVELOPMENT OF ENGINE
C       BENEVAL(N,6)=TECHNICAL DESIRABILITY
C       BENEVAL(N,7)=GIMBAL CAPABILITY
C       BENEVAL(N,8)=STAGE LENGTH
C       BENEVAL(N 9)=FABRICABILITY
C       BENEVAL(N,10)=REPAIRABILITY (IN ORBIT)
C      CAPBEN-MISSION CAPTURE BENEFIT RATING
C      CLANCOS-TOTAL COMMERCIAL PAYLOAD LAUNCH COST
C      COSRAT-LIFE CYCLE COST PER STAGE PER BENEFIT POINT
C      COUNTER-COUNTS THRUST ITERATIONS FOR PAGE FORMAT
C      ENGTYP- CODE FOR ENGINE TYPE  1=UPRATED RL10, 2=ADVANCED
C              ENGINE, 3=DEDICATED LOW THRUST ENGINE
C      EPS-ELECTRICAL POWER SUPPLY (WEIGHT(LBS) X POWER LEVEL(WATTS))
C      EPSRDT-ELECTRICAL POWER SUPPLY RDT AND E COST
C      EPSUNIT-ELECTRICAL POWER SUPPLY UNIT COST
C      ERROR-PERCENTAGE ERROR OF ACCELERATION BETWEEN APPROXIMATED AND
C            TRUE VALUE
C      ETOTAL-TOTAL NUMBER OF ENGINES TO BE PRODUCTED
C              FROM LEARNING CURVE
C      FED-FEED AND DUMP SYSTEM MASS (KG)
C      FEDRDT-FEED AND DUMP SYSTEM RDT AND E COST
C      FEDUNIT-FEED AND DUMP SYSTEM UNIT COST
C      FEDWT-FEED AND DUMP SYSTEM WEIGHT (LBS)
C      FUNIT-FIRST STAGE UNIT COST
C      GLANCOS-TOTAL GOVERNMENT PAYLOAD LAUNCH COST
C      LANCOS-TOTAL LAUNCH COST IN MILLIONS OF DOLLARS
C      LCC-LIFE CYCLE COST
C      LHRDT-LIQUID HYDROGEN TANK RDT AND E COST
C      LHTNK-LIQUID HYDROGEN TANK MASS (KG)
C      LHTNKWT-LIQUID HYDROGEN TANK WEIGHT (LBS)
C      LHUNIT-LIQUID HYDROGEN TANK UNIT COST
C      LORDT-LOX TANK RDT AND E COST
C      LOTNK-LOX TANK MASS (KG)
C      LOTNKWT-LOX TANK WEIGHT (LNBS)
C      LOUNIT-LOX TANK UNIT COST
C      LREDDT-LIQUID ROCKET ENGINE RDT&E COST
C      LREUNIT-LIQUID ROCKET ENGINE UNIT COST
C      MISCAP-MISSION CAPTURE RATING
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C     MISMOD=A N X 6 MATRIX CONTAINING MISSION INFORMATION
C       MISMOD(N,1)=MISSION NUMBER
C       MISMOD(N,2)=MOST CONSERVATIVE THRUST OF MISSION ACCELERATION
C                   RANGE, NEWTONS
C       MISMOD(N,3)=LEAST CONSERVATIVE THRUST OF MISSION ACCELERATION
C                   RANGE, NEWTONS
C       MISMOD(N,4)=MINIMUM THRUST TO DELIVER PAYLOAD, NEWTONS
C       MISMOD(N 5)=MISSION CAPTURE INDEX SLOPE
C       MISMOD(N,6)=MISSION CAPTURE INDEX INTERCEPT
C      MISSION-AN N BY 8 MARTIX CONTAINING MISSION MODEL INPUTS
C       MISSION(N,1)-MISSION NUMBER
C       MISSION(N 2)-PAYLOAD WEIGHT (KG)
C       MISSION(N 3)-MOST CONSERVATIVE ACCELERATION (G)
C       MISSION(N,4)-LEAST CONSERVATIVE ACCELERATION (G)
C       MISSION(N,5)-MISSION PROBABILITY
C       MISSION(N,6)-NUMBER OF STAGES
C       MISSION(N,7)-TYPE OF STAGE  1 FOR GOVERNMENT  2 FOR COMMERCIAL
C      N-MISSION NUMBER
C      NGOVERN-NUMBER OF GOVERNMENT LAUNCHES
C      NSTAG-TOTAL NUMBER OF STAGES (INTEGER)
C      NSTAGES-TOTAL NUMBER OF GOVERNMENT AND COMMERCIAL STAGES
C      NUMCOM-NUMBER OF COMMERCIAL LAUNCHES
C      OPSCOST-OPERTIONAL COST
C      PPSRDT-PRIMARY PROPULSION SYSTEM RDT&E COST
C      PPSUNIT-PRIMARY PROPULSION SYSTEM FIRST UNIT COST
C      PRESRDT-PRESSURIZATION SYSTEM RDT AND E COST
C      PRESS-PRESSURIZATION SYSTEM MASS (KG)
C      PRESSWT-PRESSURIZATION SYSTEM WEIGHT (LBS)
C      PROCOST-PRODUCTION COST
C      PRSUNIT-PRESSURIZATION SYSTEM UNIT COST
C      PSI-MISSION CAPTURE INDEX
C      RATLEN-STAGE LENGTH RATING
C      RDTE-RDT AND E COST OF HARDWARE
C      SEMRDT-SYSTEM ENGINEERING MANAGEMENT RDT AND E COST
C      SMIUNIT-SYSTEM MANAGEMENT INTEGRATION TEST UNIT COST
C      STAGLEN-STAGE LENGTH
C      STOTSL-TOTAL NUMBER OF STAGES TO BE PRODUCED
C             FROM LEARNING CURVE
C      STR-STAGE STRUCTURE MASS
C      STRRDT-STAGE ASTRUCTURE RDT AND E COST
C      STRUNIT-STAGE STRUCTURE UNIT COST
C      STRWT-STAGE STRUCTURE WEIGHT (LBS)
C      SUPCOS-SUPPORT COST
C      T-THRUST APPROXIMATION
C      TCP-THERMAL CONTROL-PASSIVE MASS (KG)
C      TCPRDT-THERMAL CONTROL-PASSIVB RDT AND E COST
C      TCPUNIT-THERMAL CONTROL-PASSIVE UNIT COST
C      TCPWT-THERMAL CONTROL-PASSIVE WEIGHT (LBS)
C      TESRDT-SYSTEM TEST RDT AND E COST
C      TH-FINAL THUST LEVEL
C      THRUST-ENGINE THRUST LEVEL
C      TINC-THRUST LEVEL INCREMENT
C      TL-INITIAL THRUST LEVEL
C      TOPS-TOTAL OPERATION COST FOR ALL MISSIONS CAPTURED
C      TOT-ADDDITIONAL NUMBER OF STAGES PRODUCTED FROM LEARNING CURVE
C      TOTC-TOTAL OPERATION COST FOR ONE MISSION
C      TRANS-TRANSFER COST OF DEPLOYMENT OF ONE SPACECRAFT
C      TRDTE-TOTAL RDT AND E COST
C      TRPTIME-TRIPTIME FOR LEO TO GEO ORBIT TRANSFER
C      TTC-TELEMETRY,TRACKING,AND COMMAND MASS (KG)
C      TTCRDT-TELEMETRY  TRACKING ,AND COMMAND RDT AND E COST
C      TTCUNIT-TELEMETRY, TRACKING, AND COMMAND UNIT COST
C      TTCWT-TELEMETRY  TRACKING  AND COMMAND WEIGHT (LBS)
C      UNIT-SUM OF HARDWARE UNIT COST
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C     WBO-BURN OUT MASS OF STAGE
C     WFINAL-SPACECRAFT BURN OUT MASS
C     WTXPL-ELECTRICAL POWER SUPPLY   WEIGHT X POWER LEVEL (LBS X WATTS)
C
C XXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXXX
C
      DIMENSION MISMOD(40,6),PSI(40) BENEVAL(2 10),MISSION(40,7)
      REAL MISCAP MISMOD NUMCOM NGOVERN,MISSION NSTAGES LCC LANCOS
     $,LOTNK LHTNK,LOTNKWT,LHTNKWT LHUNIT,LOUNIT LREUNIT,LHRDT,LORDT
     $,LREDDT
      INTEGER ENGTYP THRUST TL,TH TINC,N,COUNTER
      REWIND 2
      REWIND 3
C READ ENGINE THRUST RANGE,THRUST INCREMENT,NUMBER OF MISSIONS AND BURNS
      READ(2 530) ENGTYP,TL,TH TINC,N
C READ ENGINE DDT&E AND FIRST UNIT COST
      READ(2 520) LREDDT,LREUNIT
      IF(ENGTYP GT 1)GO TO 10
      WRITE(3,501)
      GO TO 20
   10 IF(ENGTYP GT 2)GO TO 15
      WRITE(3,502)
      GO TO 20
   15 WRITE(3,503)
C READ IN MISSION  INFORMATION
   20 READ(2,560) ((MISSION(I,J),J=1,7),I=1,N)
C READ IN BENEFIT EVALUATION MATRIX
      READ(2,570)((BENEVAL(I,J) J=1,10),I=1 2)
C       READ IN MASSES FOR COST CALCULATION
      READ(2,580)TTC,ACS EPS,LHTNK LOTNK,STR,FED,PRESS,TCP
      WRITE(3,506) TL,TH,TINC
      WRITE(3,545)
      WRITE(3,546)((MISSION(I,J),J=1,7),I=1,N)
      WRITE(3,507)
      WRITE(3,508)((BENEVAL(I,J),J=1 5),I=1,2)
      WRITE(3,509)((BENEVAL(I,J),J=6,10),I=1,2)
      WRITE(3,547)
      WRITE(3,548)TTC ACS,EPS,LHTNK LOTNK,STR,FED PRESS TCP
C
C        CONVERT KG TO LBS FOR COST EQUATIONS
C
      TTCWT=2 2046*TTC
      ACSWT=2 2046*ACS
      WTXPL=2 2046*EPS
      LHTNKWT=2 2046*LHTNK
      LOTNKWT=2 2046*LOTNK
      STRWT=2 2046*STR
      FEDWT=2 2046*FED
      PRESSWT=2 2046*PRESS
      TCPWT=2 2046*TCP
C
C
C                  RDT AND E COSTS
C
C       TELEMETRY, TRACKING, AND COMMAND
      TTCRDT=1188 68+54 81*TTCWT
C       ATTITUDE CONTROL
      ACSRDT=1494 78+98 61*(ACSWT**0 81)
C       ELECTRICAL POWER SUPPLY
      EPSRDT=2648 8+0 031*(WTXPL**0 97)
C       PROPELLANT TANKS
      LHRDT=3869 8*(LHTNKWT**0 13)
      LORDT=9674 5*(LOTNKWT**0 13)
C       STRUCTURE
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      STRRDT=754 9+70 8*(STRWT**O 66)
C       FEED AND DUMP
      FEDRDT=1382 O*(FEDWT**O 36)
C       PRESSURIZATION SYSTEM
      PRESRDT=3289 O*(PRESSWT**O 21)
C       THERMAL CONTROL - PASSIVE
      TCPRDT=251 62+29 46*(TCPWT**O 66)
C       SUM RDT AND E COSTS
      RDTE=TTCRDT+ACSRDT+EPSRDT+LHRDT+LORDT+STRRDT+FEDRDT+PPESRDT
     $+TCPRDT
C       SYSTEM ENGINNERING MANAGEMENT
      SEMRDT=O 25*RDTE+O 25*LREDDT
C       SYSTEMS TEST
      TESRDT=O 45*RDTE+O 45*LREDDT
C       TOTAL RDT AND E COSTS IN MILLIONS
      TRDTE=(RDTE+SEMRDT+TESRDT)/1000 O
      PPSRDT=LREDDT+TRDTE
C
C
C
C                  UNIT COSTS
C
C
C       TELEMETRY  TRACKING, AND COMMAND
      TTCUNIT=51 34+36 94*(TTCWT**O 93)
C       ATTITUDE CONTROL
      ACSUNIT=17 59*(ACSWT**O 69)
C       ELECTRICAL POWER SUPPLY
      EPSUNIT=66 72*(WTXPL**O 29)
C       PROPELLANT TANKS
      LHUNIT=7 91*(LHTNKWT**O 68)
      LOUNIT=15 8*(LOTNKWT**O 68)
C       STRUCTURE
      STRUNIT=2 51*(STRWT**O 96)
C       FEED AND DUMP
      FEDUNIT=114 O+O 08*FEDWT
C       PRESSURIZATION SYSTEM
      PRSUNIT=157 O+O 42*(PRESSWT**O 77)
C       THERMAL CONTROL - PASSIVE
      TCPUNIT=4 25*(TCPWT**O 65)
C       SUM UNIT COSTS
      UNIT=TTCUNIT+ACSUNIT+EPSUNIT+LHUNIT+LOUNIT+STRUNIT+FEDUNIT+
     $PRSUNIT+TCPUNIT
C       SYSTEMS MANAGEMENT INTEGRATION TEST
      SMIUNIT=O 30*UNIT+O 30*LREUNIT
C       FIRST UNIT COST IN MILLIONS
      FUNIT=(UNIT+SMIUNIT)/1000 O
      PPSUNIT=LREUNIT+FUNIT
C
C
C       BEGIN COST AND BENEFIT CALCULATION
C
C
C       CALCULATE MOST CONSERVATIVE AND LEAST CONSERVATIVE THRUSTS
C       FOR EACH MISSION
C
      DO 46 J=3 4
      DO 45 I=1 N
C
C       GUESS INITIAL THRUST
      T=(TH+TL)/2 O
C
C       CALCULATE BURNOUT WEIGHT OF STAGE
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```
   25 IF(ENGTYP GT 1) GO TO 30
      WBO=3079 1*T**(-0 0075)
      GO TO 40
   30 IF(ENGTYP GT 2) GO TO 35
      WBO=2980 0*T**(-0 0062)
      GO TO 40
   35 WBO=2876 3*T**(-0 0087)
C
C      CALCULATE SPACECRAFT BURNOUT WEIGHT
C
   40 WFINAL=(WBO+MISSION(I,2))
C      APPROXIMATE FINAL ACCELERATION LEVEL
      ACCEL=T/(WFINAL*9 81)
C      CALCULATE FINAL ACCELERATION ERROR
      ERROR=(ABS(MISSION(I,J)-ACCEL))/MISSION(I J)
      IF(ERROR LT 0 00001) GO TO 42
      T=MISSION(I,J)*WFINAL*9 81
      GO TO 25
   42 K=J-1
   45 MISMOD(I,K)=T
   46 CONTINUE
C
C      CALCULATE MINIMUM THRUST TO DELIVER PAYLOAD
C
      DO 65 I=1 N
      IF(ENGTYP GT 1) GO TO 55
      MISMOD(I,4)=(MISSION(I 2)/3705 8)**(14 92537)
      GO TO 64
   55 IF(ENGTYP GT 2) GO TO 60
      MISMOD(I,4)=(MISSION(I,2)/4666 7)**(19 60784)
      GO TO 64
   60 MISMOD(I,4)=(MISSION(I,2)/4217 6)**(13 88888)
C
C      CALCULATE MISSION CAPTURE INDEX SLOPE AND INTERCEPT
C
   64 MISMOD(I,5)=-1 0/(MISMOD(I,3)-MISMOD(I 2))
   65 MISMOD(I,6)=-1 0*(MISMOD(I,3)*MISMOD(I,5))
C
C       TRANSFER MISSION NUMBERS TO MISMOD MATRIX
      DO 70 I=1,N
   70 MISMOD(I 1)=MISSION(I,1)
C
C
C      WRITE MISSION MODEL MATRIX
      WRITE(3 504)
      WRITE(3,505)
      WRITE(3,540)((MISMOD(I J),J=1 6),I=1,N)
C      WRITE COST STATEMENTS
      WRITE(3 553)
      WRITE(3,525) LREDDT,LREUNIT
      WRITE(3 554)TRDTE,FUNIT
      WRITE(3,555)PPSRDT PPSUNIT
C
C      BEGIN THRUST INTERATION
C
      IF( NOT (ENGTYP GT 1))GOTO 85
      IF( NOT (ENGTYP GT 2))GOTO 75
      WRITE(3 503)
      GOTO 80
C     ELSE
   75 WRITE(3 502)
   80 CONTINUE
C     ENDIF
      GOTO 87
```

```
C       ELSE
   85 WRITE(3,501)
   87 CONTINUE
C       ENDIF
      WRITE(3,553)
      WRITE(3,585)
      COUNTER=1
      DO 500 THRUST=TL,TH,TINC
C CALCULATE MISSION CAPTURE INDEX BY LOCATING THRUST
      DO 130 I=1,N
      IF(THRUST GT MISMOD(I,3)) GO TO 100
      IF(MISMOD(I,2) GE MISMOD(I,4)) GO TO 90
      IF(THRUST GE MISMOD(I,4)) GO TO 120
      GO TO 100
   90 IF(THRUST GT MISMOD(I,2)) GO TO 120
      IF(THRUST GE MISMOD(I,4)) GO TO 110
  100 PSI(I)=0 0
      GO TO 130
  110 PSI(I)=1 0
      GO TO 130
  120 PSI(I)=MISMOD(I,5)*THRUST+MISMOD(I,6)
  130 CONTINUE
C
C
C        NUMBER OF GOVERNMENT AND COMMERCIAL STAGES
C
C        SET NUMBER OF STAGES EQUAL TO ZERO
      NUMCOM=0 0
      NGOVERN=0 0
      DO 170 I=1,N
      IF( NOT (PSI(I) GT 0 0))GOTO 160
      IF( NOT (MISSION(I,7) GT 1))GOTO 140
      NUMCOM=NUMCOM+MISSION(I,6)
      GOTO 150
C       ELSE
  140 NGOVERN=NGOVERN+MISSION(I,6)
  150 CONTINUE
C       ENDIF
  160 CONTINUE
C       ENDIF
  170 CONTINUE
C
C        CALCULATE LAUNCH COSTS
C
      CLANCOS=55700 0*NUMCOM
      GLANCOS=31300 0*NGOVERN
C
C        TOTAL LAUNCH COSTS IN MILLIONS
C
      LANCOS=(CLANCOS+GLANCOS)/1000 0
C
C        DEPLOYMENT OPERATION COSTS
C
      TOTC=0 0
      DO 300 I=1,N
      IF(PSI(I) LE 0 0)GOTO 300
C
C        DETERMINE STAGE BURNOUT MASS
C
      IF(ENGTYP GT 1)GO TO 190
      WBO=3079 1*THRUST**(-0 0075)
      GO TO 200
  190 IF (ENGTYP GT 2)GO TO 195
      WBO=2980 0*THRUST**(-0 0062)
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```
      GO TO 200
  195 WBO=2876 3*THRUST**(-0 0087)
C         CALCULATE SPACECRAFT BURNOUT MASS
  200 WFINAL=(WBO+MISSION(I,2))
C         FINAL ACCELERATION
      ACCEL=THRUST/(WFINAL*9 81)
C         TRIPTIME CALCULATION
      IF( NOT (ACCEL GT 0 012))GOTO 280
      IF( NOT (ACCEL GT 0 017))GOTO 270
      IF( NOT (ACCEL GT 0 030))GOTO 260
      TRPTIME=-1 735*ACCEL+29 50
      GOTO265
C     ELSE
  260 TRPTIME=-345 6*ACCEL+39 9
  265 CONTINUE
C     ENDIF
      GOTO 275
C     ELSE
  270 TRPTIME=-1288 5*ACCEL+55 9
  275 CONTINUE
C     ENDIF
      GOTO 285
C     ELSE
  280 TRPTIME=-5000 0*ACCEL+100 33
  285 CONTINUE
C     ENDIF
C       TRANSFER COST FOR ONE SPACECRAFT
C
      TRANS=((TRPTIME+42 0)*1 43)/1000 0
C         TOTAL OPERATIONS COST FOR ONE MISSION
      TOPS=TRANS*MISSION(I,6)
C
C       TOTAL OPERATIONS COST FOR MISSION MODEL
C
  295 TOTC=TOTC+TOPS
  300 CONTINUE
C
C       TOTAL SUPPORT COST IN MILLIONS
      SUPCOS=LANCOS+TOTC
C
C       TOTAL PRODUCTION COSTS
C
      NSTAGES=NUMCOM+NGOVERN
      IF(NSTAGES LT 1 0) GO TO 405
      STOTAL=0 0
      ETOTAL=0 0
      NSTAG=NSTAGES
      DO 340 L=1,NSTAG
      IF(ENGTYP GT 1)GOTO 310
      ETOTAL=NSTAGES
      GOTO 330
  310 IF(ENGTYP GT 2) GOTO 320
      ENGTOT=L**(-0 1203)
      ETOTAL=ETOTAL+ENGTOT
      GO TO 330
  320 ENGTOT=L**(-0 1203)
      ETOTAL=ETOTAL+ENGTOT
  330 TOT=L**(-0 152)
  340 STOTAL=STOTAL+TOT
      PROCOST=STOTAL*FUNIT+ETOTAL*LREUNIT
C
C       AVERAGE PRODUCTION COST IN MILLIONS
      APROCOS=PROCOST/NSTAGES
C
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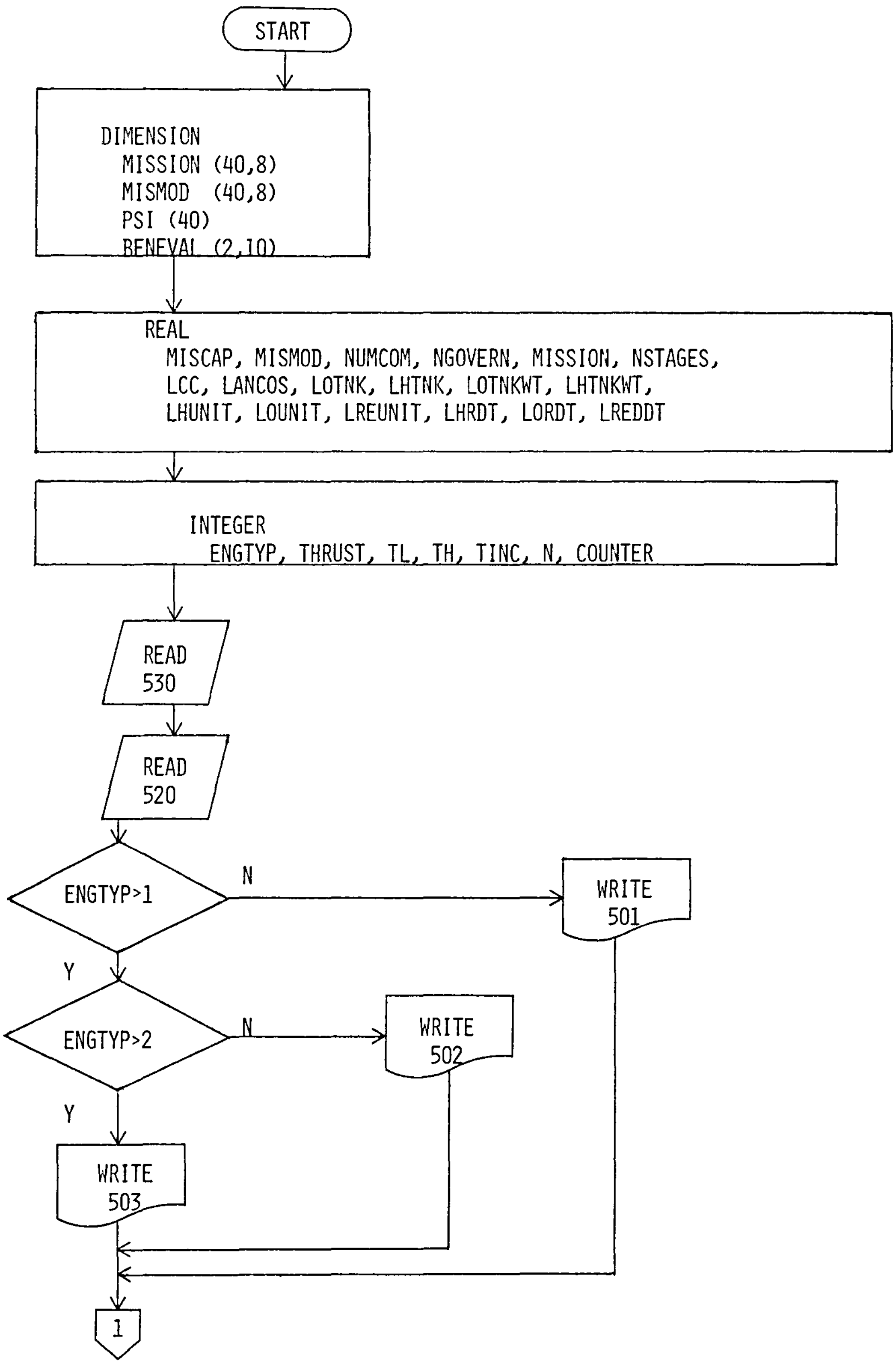
```
C       TOTAL LIFE CYCLE COST
        LCC=TRDTE+PROCOST+SUPCOS+LREDDT
C
C CALCULATE OMEGA, MISSION CAPTURE RATING
        DO 360 K=1,N
        X=(PSI(K)-1 0)*MISSION(K 5)*MISSION(K,6)
        IF(K GT 1) GO TO 350
        SUM1=0 0
        SUM2=0 0
  350   SUM1=X+SUM1
        SUM2=MISSION(K,6)+SUM2
  360   CONTINUE
        MISCAP=10 0*(1 0+(SUM1/SUM2))
        IF( NOT (ENGTYP GT 1))GOTO 375
        IF( NOT (ENGTYP GT 2))GOTO 365
        STAGLEN=6 3699*THRUST**(-0 021558)
        GOTO 370
C       ELSE
  365   STAGLEN=6 002961*THRUST**(-0 009578)
  370   CONTINUE
C       ENDIF
        GOTO 380
C       ELSE
  375   STAGLEN=7 31733*THRUST**(-0 02793)
  380   CONTINUE
C       ENDIF
        IF( NOT (BENEVAL(1,1) GT 0 0))GOTO 385
        CAPBEN=BENEVAL(1 1)
        GOTO 390
C       ELSE
  385   CAPBEN=MISCAP
  390   CONTINUE
C       ENDIF
        IF( NOT (BENEVAL(1,8) GT 0 0))GOTO 395
        RATLEN=BENEVAL(1 8)
        GOTO 400
C       ELSE
  395   RATLEN=-3 2808*STAGLEN+25 0
  400   CONTINUE
C       ENDIF
        BENEFIT=(CAPBEN*BENEVAL(2,1)+BENEVAL(1,2)*BENEVAL(2,2)
       $+BENEVAL(1,3)*BENEVAL(2 3)+BENEVAL(1,4)*BENEVAL(2 4)
       $+BENEVAL(1,5)*BENEVAL(2,5)+BENEVAL(1 6)*BENEVAL(2 6)
       $+BENEVAL(1,7)*BENEVAL(2,7)+RATLEN*BENEVAL(2 8)
       $+BENEVAL(1,9)*BENEVAL(2 9)+BENEVAL(1,10)*BENEVAL(2,10))/100 0
C
C
        COSRAT=(LCC/NSTAGES)/BENEFIT
C
C
  405   COUNTER=COUNTER+1
        IF(COUNTER LT 52)GO TO 430
        IF( NOT (ENGTYP GT 1))GOTO 420
        IF( NOT (ENGTYP GT 2))GOTO 410
        WRITE(3 503)
        GOTO 415
C       ELSE
  410   WRITE(3,502)
  415   CONTINUE
C       ENDIF
        GOTO 425
C       ELSE
  420   WRITE(3,501)
  425   CONTINUE
```

```
C      ENDIF
       WRITE(3,553)
       WRITE(3,585)
       COUNTER=1
   430 IF(NSTAGES LT  1 O) GOTO 450
       WRITE(3 590)THRUST MISCAP PROCOST NSTAGES APPOCOS LANCOS TOTC
      $SUPCOS,BENEFIT,LCC COSRAT
       GO TO 500
   450 WRITE(3,600)THRUST
   500 CONTINUE
   501 FORMAT("1",40X "COST/BENEFITS OF UPRATFD PL-10 ENGINE')
   502 FORMAT("1",44X,"COST/BENEFITS OF ADVANCED ENGINE')
   503 FORMAT("1" 38X,"COST/BENEFITS OF DEDICATED LOW THRUST FNGINE")
   504 FORMAT(////48X,"MISSION MODEL MATRIX ",//)
   505 FORMAT(2X,"MISSION" 8X,"MOST" 11X,"LEAST" 8X "MIN  THRUST",3X,
      $"MISSION CAPTURE INDEX" / 2X "NUMBER"
      $5X,"CONSERVATIVE" 3X,"CONSERVATIVE" 4X "TO DELIVER" 6X
      $"SLOPE   INTERCEPT" /,13Y
      $"THRUST N",9X,"THRUST,N",7X,"PAYLOAD, N"/)
   506 FORMAT(//10X,"INITIAL THRUST= ' I5 1X,'NEWTONS"10Y
      $"FINAL THRUST= ",I5,1X,"NEWTONS",10X "THRUST INCREMENT= "
      $,I5 1X,"NEWTONS')
   507 FORMAT(/////,48X "BENEFIT INPUT MATRIY' / )
   508 FORMAT(20X,"MISSION" 9X,"RELIABILITY',9Y 'TECHNICAL" 10X "GROWTH"
      $12X,"LENGTH OF" / 20X,"CAPTURE',10X "ENGINE"
      $15X,"RISK",12X,"POTENTIAL"
      $9X,"DEVELOPMENT",// 2X,"RATING",13X,F5 2 13X F5 2 15Y F5 2 13X,
      $F5 2,13X,F5 2,//,2X "WEIGHTING",9X F5 1 13X F5 1,15X F5 1
      $13X,F5 1,13X,F5 1,/,2X,"FACTOR',//)
   509 FORMAT(22X "TECHNICAL" 5X  RELIABILITY" 11X
      $"STAGE",9X,"FABRICABILITY"
      $ 7X,"REPAIRABILITY" /,20X 'DESIRABILITY" 7X, STAGE ,14X,
      $"LENGTH" 31X,"(IN ORBIT)" '/ 2X "RATING" 13X,F5 2,13X F5 2,15X
      $F5 2,13X,F5 2,13X,F5 2,//,2X,"WEIGHTING" 9Y F5 1 13X F5 1,15X,
      $F5 1,13X F5 1,13X F5 1,/ 2Y,"FACTOR",//)
   520 FORMAT(10X E10 2 10X E9 2)
   525 FORMAT(// 15X " LIQUID ROCKET ENGINE RDT&E COST= "
      $1PE10 2,10X "LIQUID ROCKET ENGINE FIRST UNIT COST= ",E9 2)
   530 FORMAT(2X I1,2Y I5 2X,I5 2X,I5 2X I2)
   540 FORMAT(2X F4 1 7X,F8 1 9X,F8 1,7X,F8 1,3X E11 3,4X F8 4)
   545 FORMAT(/// 50X "MISSION INFORMATION",// 6Y "MISSION" 6X "PAYLOAD",
      $12X,"MOST" 17X "LEAST",12X "PROBABILITY" 6Y,'NUMBER" 6X,"TYPE",/
      $6X,"NUMBER",7X,"WEIGHT",9X,"CONSERVATIVE",10X,"CONSERVATIVE",
      $27X,"OF",9X,"OF" / 21X,"(KG)" 9X,"ACCELERATION (G)",6Y
      $"ACCELERATION (G)" 21X,"STAGES",6X,"STAGE" /)
   546 FORMAT(7X F4 1 8X,F7 1,12X F5 3,16X F5 3 16X F4 2 10X,F4 1,
      $8X,F3 1)
   547 FORMAT("1",/// 43X "PROPULSION SUBSYSTEM MASSES (KG)",//
      $,4X "TELEMETRY,",3X,"ATTITUDE" 4X,"ELECTRICAL",4X
      $"PROPELLANT",3X "STRUCTURE" 5X,"FEED AND" 8X,
      $"PRESSURIZATION" 3X,"PASSIVE",/ 4X,
      $"TRACKING " 4X "CONTROL",4X "POWER SUPPLY",6X,"TANK" 8X,'MASS",
      $7X,"DUMP SYSTEM" 10X "SYSTEM" 7X "THERMAL" / 3X,
      $"AND COMMAND" 4X "MASS",6X,"(KG X WATTS)",6X "MASS",
      $21X,"MASS" 16X,"MASS",8X "CONTROL",/,6X,
      $"MASS",34X,"LH2  LO2" 53X "MASS" /)
   548 FORMAT(6X,F5 1 7X,F5 1,6X,F8 1 5X,F5 1 2X F5 1,4X,F5 1,
      $8X,F5 1,15Y F5 1,9X F5 1)
   550 FORMAT(12X,F5 3)
   553 FORMAT(/// 34X "ALL COSTS ARE IN MILLIONS OF 1982 FISCAL YEAR" 1X
      $"DOLLARS")
   554 FORMAT(//,5X,"STAGE WITHOUT ENGINE TOTAL RDT AND E COST= "
      $0PF8 3,10X "STAGE WITHOUT ENGINE FIRST UNIT COST= ",F8 3)
   555 FORMAT(// 5X "PRIMARY PROPULSION SYSTEM RDT&E COST =" F8 3 10X
```

```
   $"PRIMARY PROPULSION SYSTEM FIRST UNIT COST =",F8 3)
560 FORMAT(3X,F4 1,3X,F7 1,3X,F5 3 3X,F5 3,3X F4 2,3X,F4 1,3X,F3 1)
570 FORMAT(10F6 2/10F6 1)
580 FORMAT(2X,F5 1,2X,F5 1,2X F8 1 2X,F5 1 2X,F5 1,2X F5 1,
   $2X,F5 1,2X,F5 1 2X F5 1)
585 FORMAT(//,3X,"THRUST",5X,"MISSION",5X "PRODUCTION" 3X,"NUMBER",3X,
   $"AVERAGE",
   $5X,"LAUNCH",5X "DEPLOYMENT",5X,"SUPPORT" 5X,"BENEFIT",6X "LCC",
   $5X,"LCC PER",/,2X,
   $"(NEWTONS)",3X,"CAPTURE",7X,"COST",9X 'OF" 6X,
   $"UNIT",8X,"COST",8X,
   $"COST",10X,"COST",27X,"LCC",4X "STAGE PER",/
   $14X,"RATING",19X,"STAGES",
   $4X,"COST",66X,"BENEFIT",/)
590 FORMAT(2X,I7,5X,F6 3,5X,F9 3,5X,F5 1,5X F6 3,3X,F9 3 2X,F9 3,
   $6X,F9 3,4X,F7 3,3X,F10 3,1X F8 3)
600 FORMAT(2X,I7,30X,"THERE ARE NO MISSIONS CAPTURED AT THIS",1X
   $,"THRUST LEVEL")
    STOP
    END
```

TABLE B-4 RACE FLOWCHART



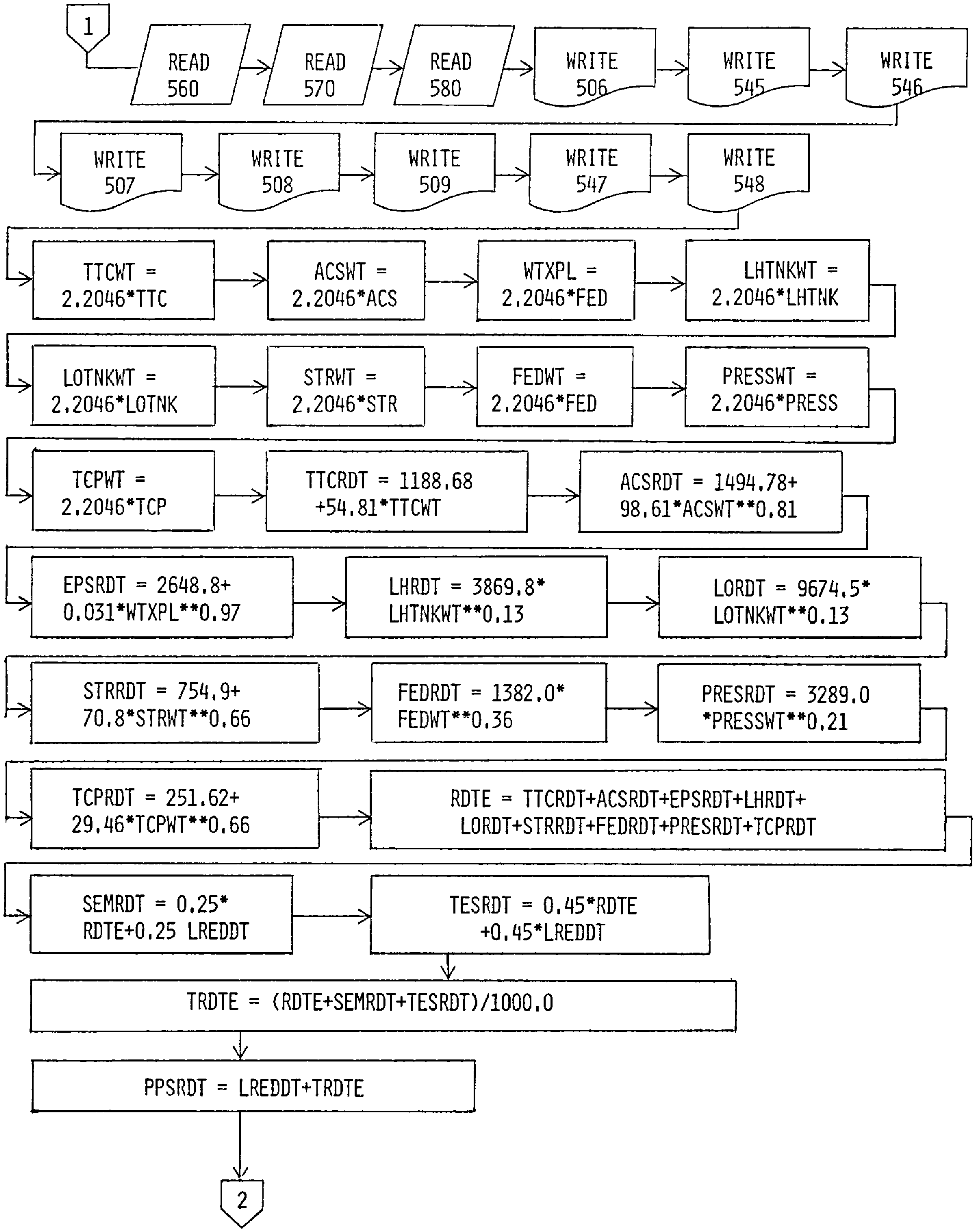

1
READ 560
READ 570
READ 580
WRITE 506
WRITE 545
WRITE 546
WRITE 507
WRITE 508
WRITE 509
WRITE 547
WRITE 548
TTCWT = 2.2046*TTC
ACSWT = 2.2046*ACS
WTXPL = 2.2046*FED
LHTNKWT = 2.2046*LHTNK
LOTNKWT = 2.2046*LOTNK
STRWT = 2.2046*STR
FEDWT = 2.2046*FED
PRESSWT = 2.2046*PRESS
TCPWT = 2.2046*TCP
TTCRDT = 1188.68 +54.81*TTCWT
ACSRDT = 1494.78+ 98.61*ACSWT**0.81
EPSRDT = 2648.8+ 0.031*WTXPL**0.97
LHRDT = 3869.8* LHTNKWT**0.13
LORDT = 9674.5* LOTNKWT**0.13
STRRDT = 754.9+ 70.8*STRWT**0.66
FEDRDT = 1382.0* FEDWT**0.36
PRESRDT = 3289.0 *PRESSWT**0.21
TCPRDT = 251.62+ 29.46*TCPWT**0.66
RDTE = TTCRDT+ACSRDT+EPSRDT+LHRDT+ LORDT+STRRDT+FEDRDT+PRESRDT+TCPRDT
SEMRDT = 0.25* RDTE+0.25 LREDDT
TESRDT = 0.45*RDTE +0.45*LREDDT
TRDTE = (RDTE+SEMRDT+TESRDT)/1000.0
PPSRDT = LREDDT+TRDTE
2

TABLE B-4 RACE FLOWCHART (CONT'D)

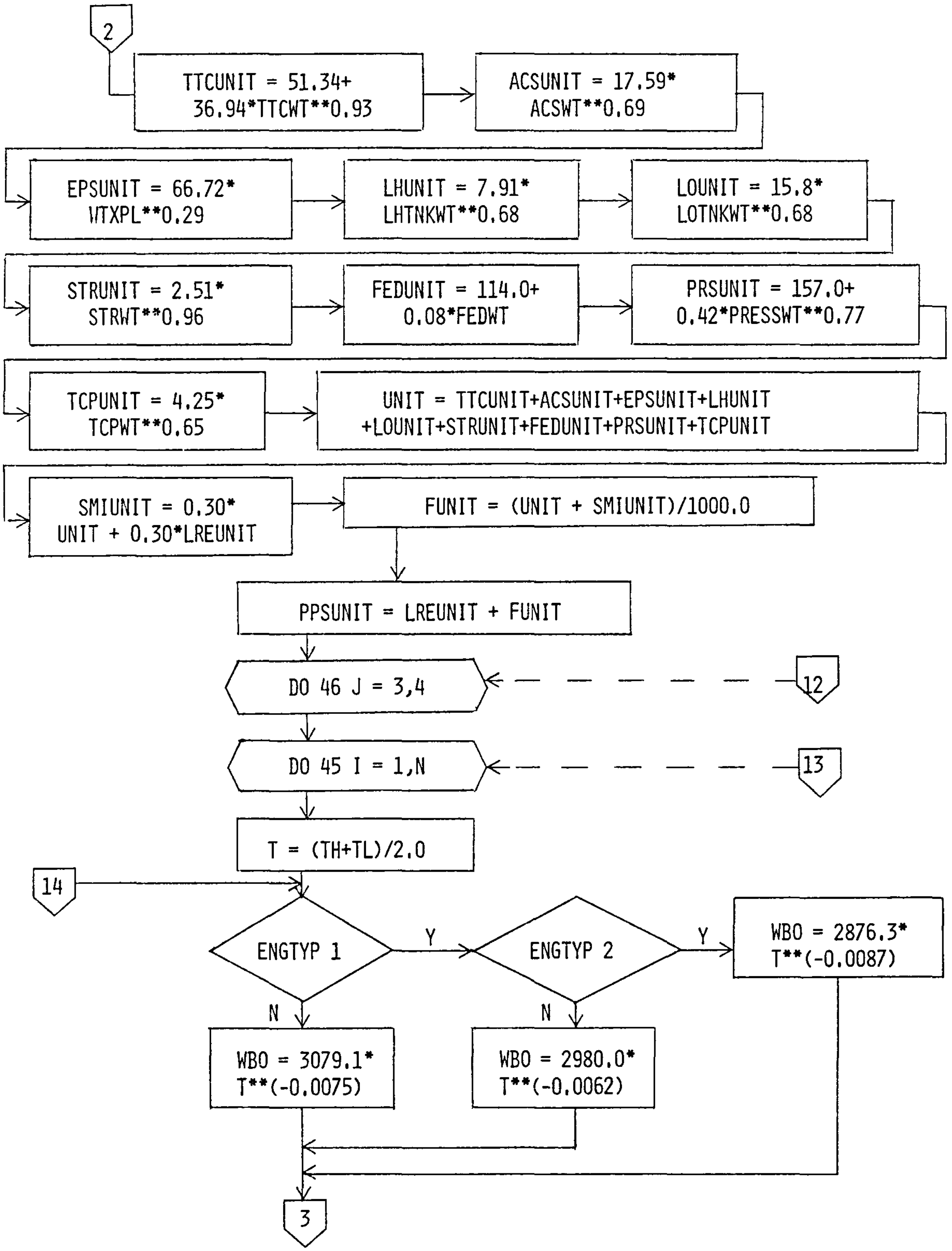


TABLE B-4 RACE FLOWCHART (CONT'D)

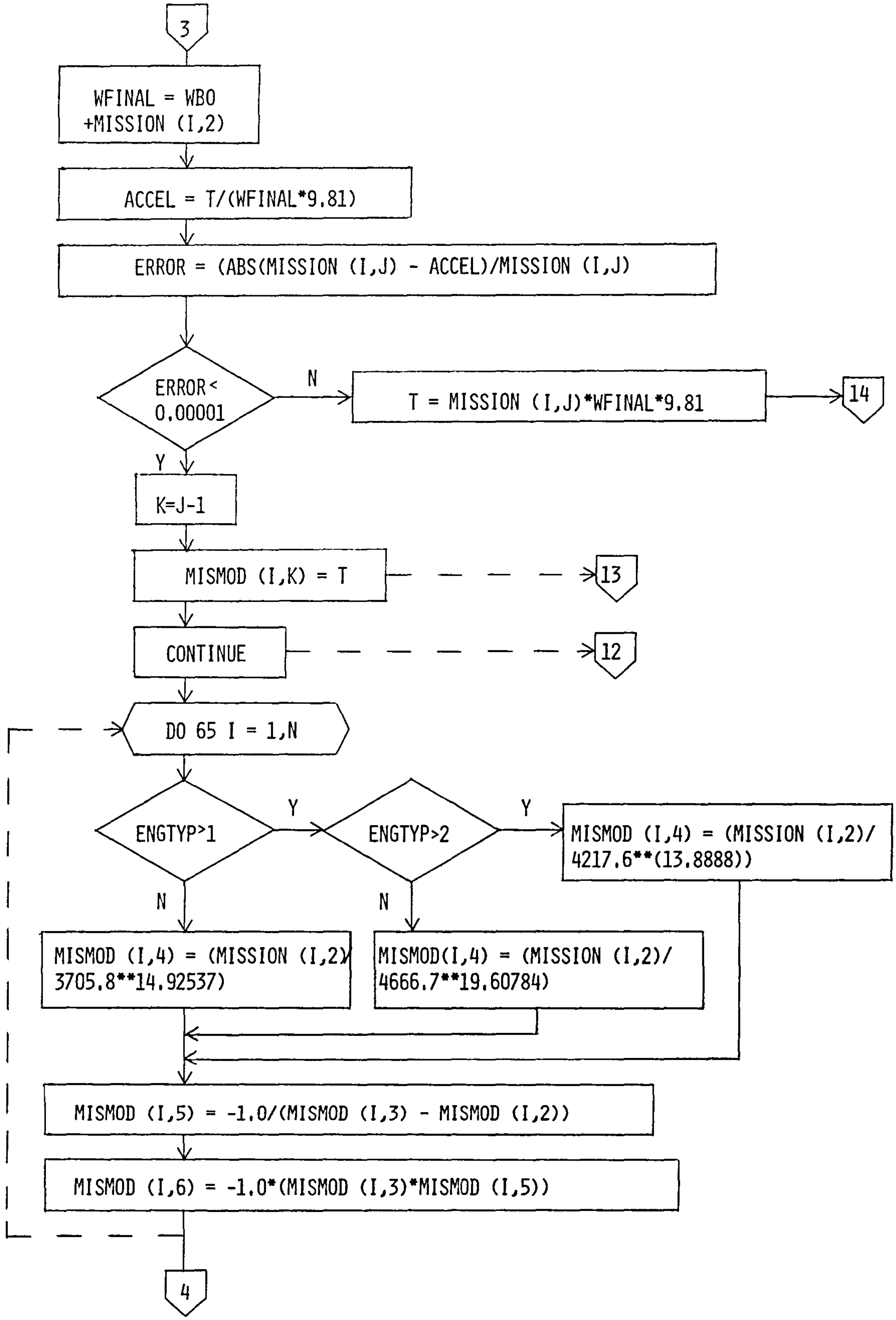


TABLE B-4 RACE FLOWCHART (CONT'D)

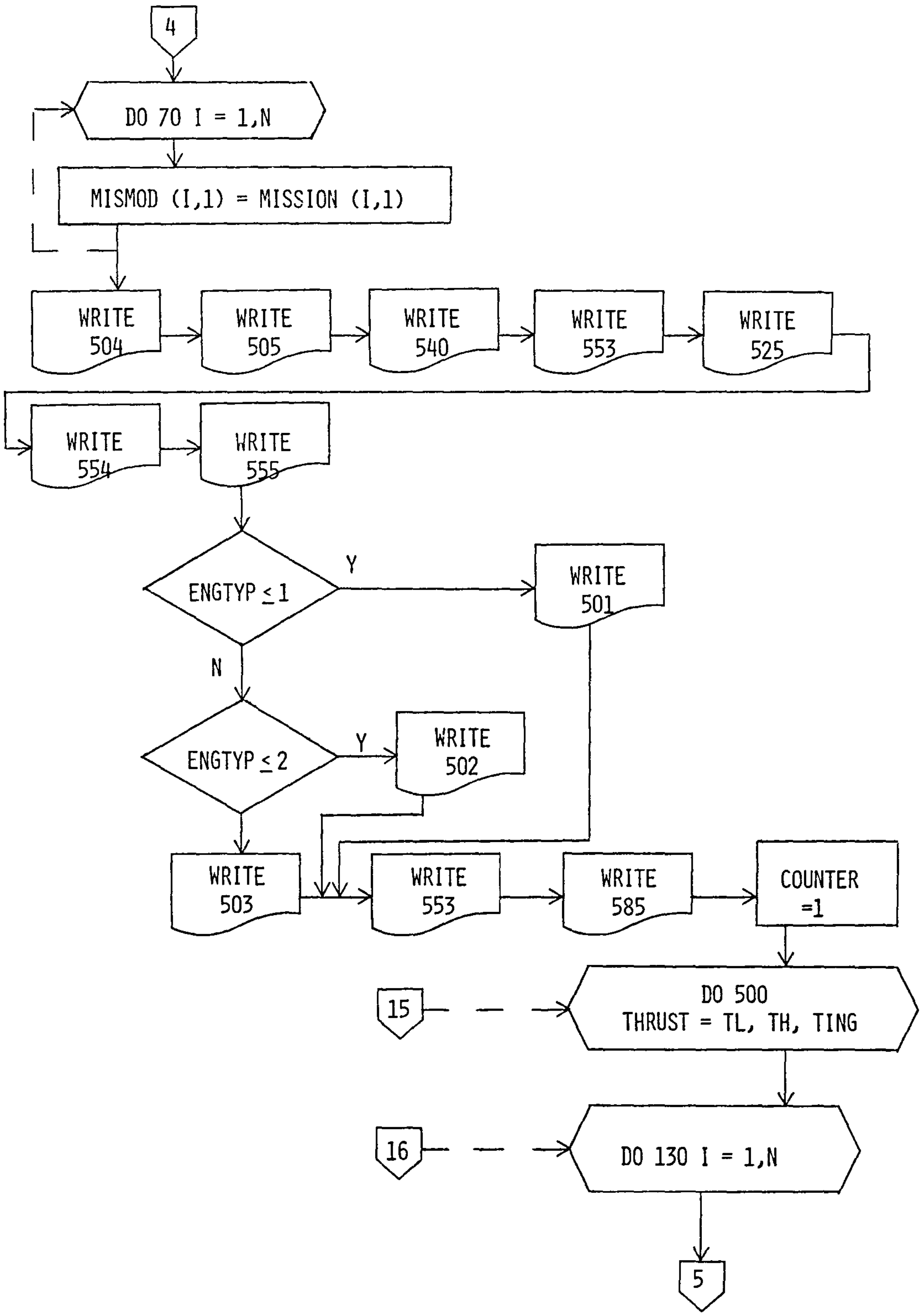


TABLE B-4 RACE FLOWCHART (CONT'D)

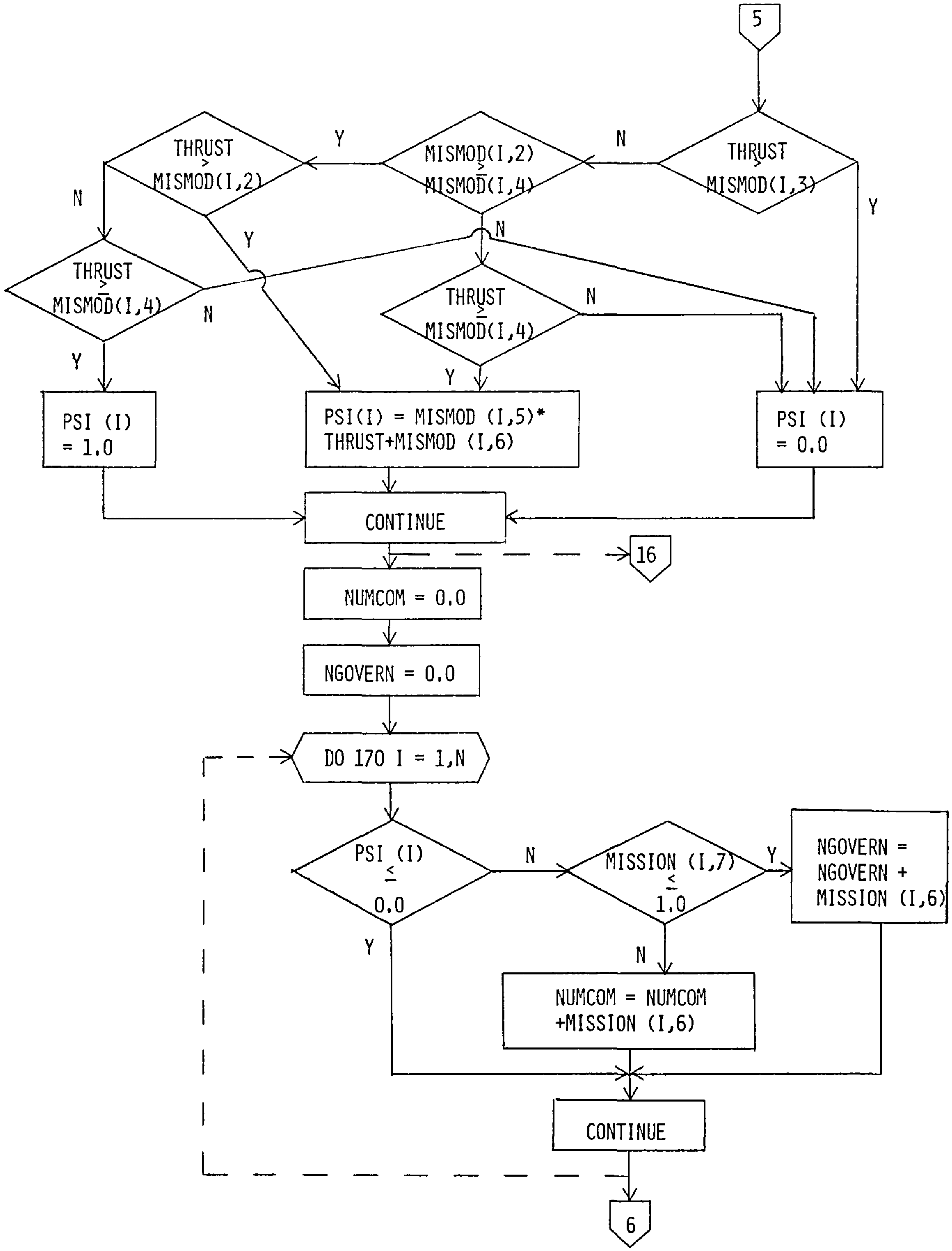


TABLE B-4 RACE FLOWCHART (CONT'D)

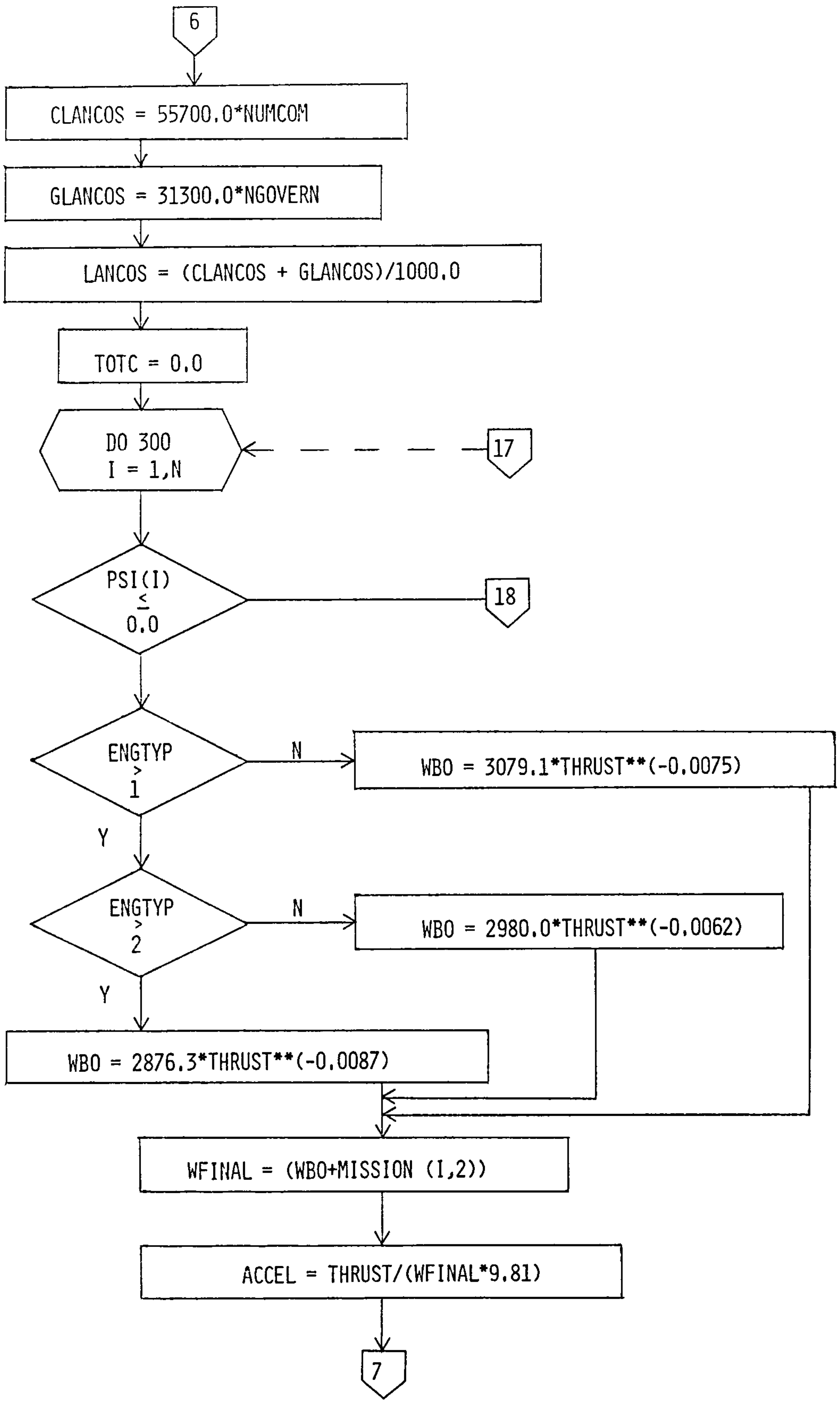
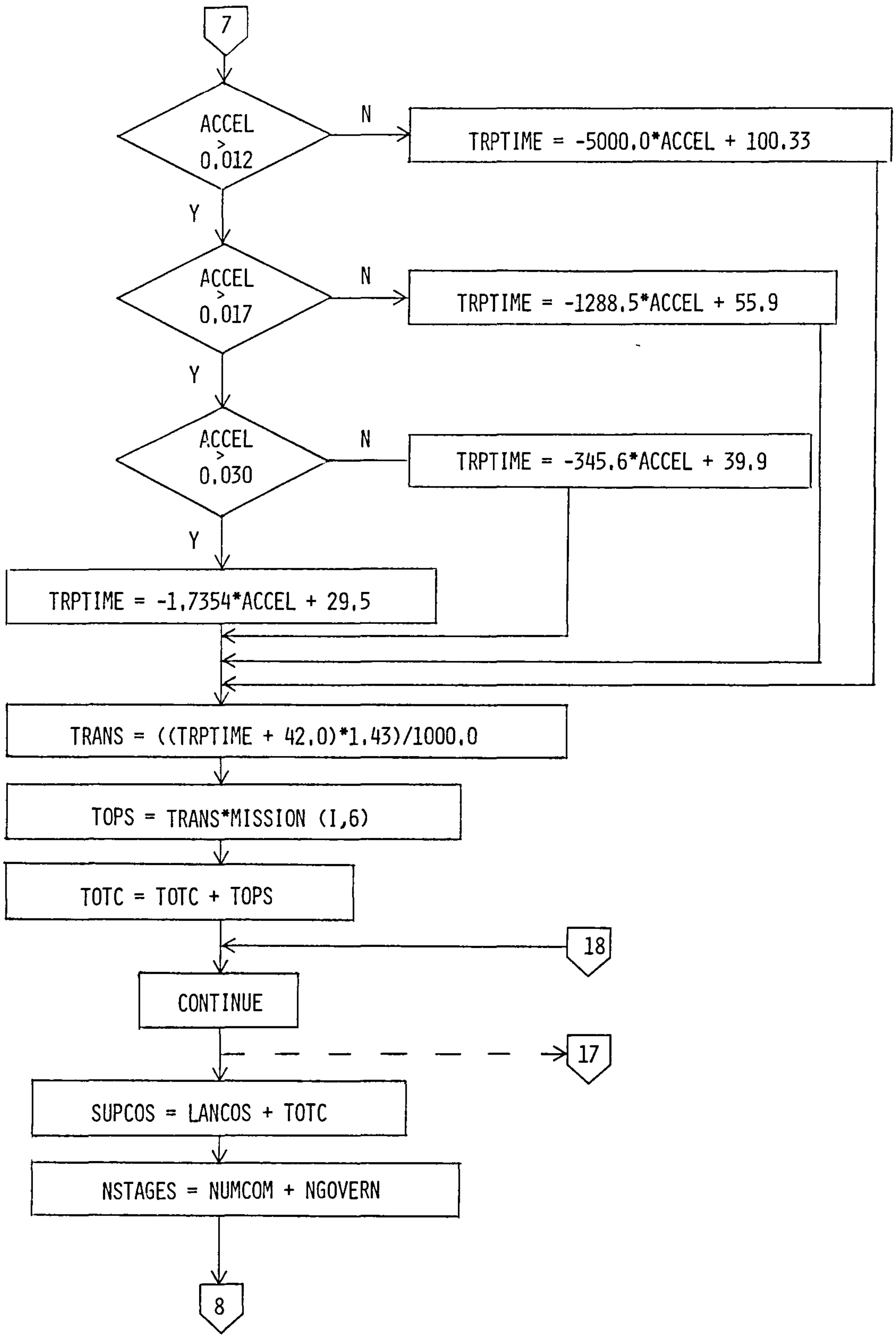
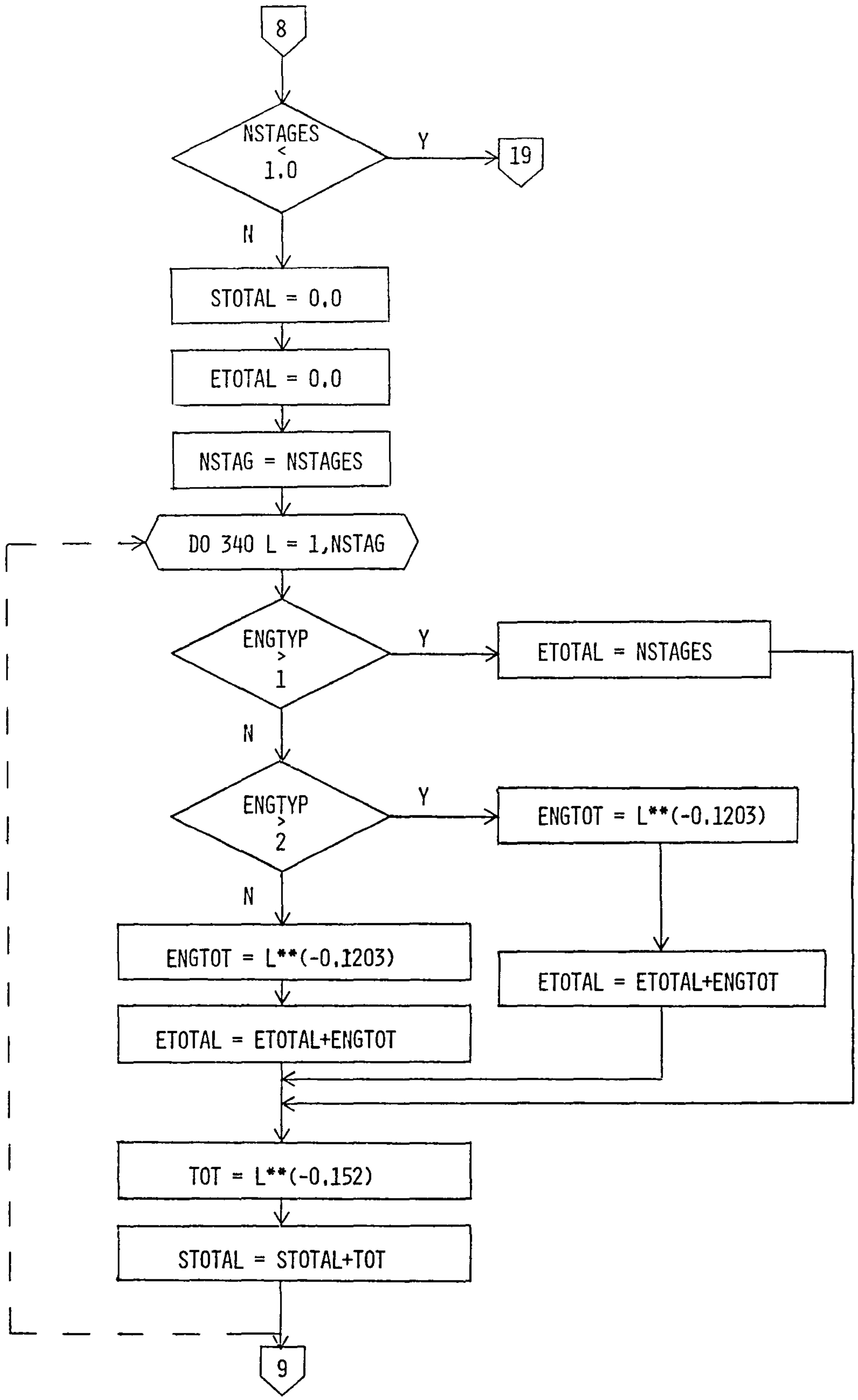
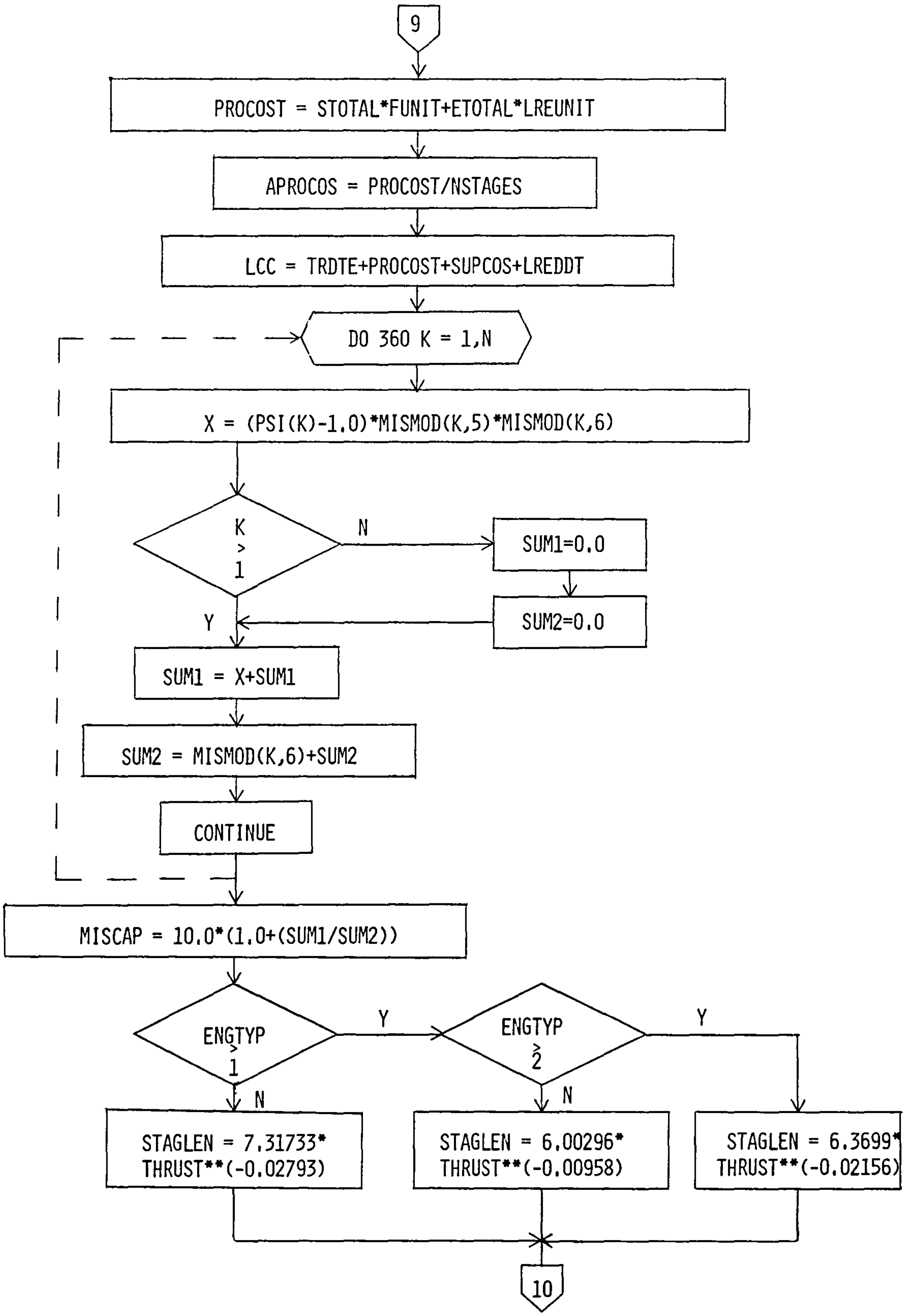


TABLE B-4 RACE FLOWCHART (CONT'D)




8
NSTAGES
<
1.0
Y
19
N
STOTAL = 0.0
ETOTAL = 0.0
NSTAG = NSTAGES
DO 340 L = 1,NSTAG
ENGTYP
>
1
Y
ETOTAL = NSTAGES
N
ENGTYP
>
2
Y
ENGTOT = L**(-0.1203)
N
ENGTOT = L**(-0.1203)
ETOTAL = ETOTAL+ENGTOT
ETOTAL = ETOTAL+ENGTOT
TOT = L**(-0.152)
STOTAL = STOTAL+TOT
9


9
PROCOST = STOTAL*FUNIT+ETOTAL*LREUNIT
APROCOS = PROCOST/NSTAGES
LCC = TRDTE+PROCOST+SUPCOS+LREDDT
DO 360 K = 1,N
X = (PSI(K)-1.0)*MISMOD(K,5)*MISMOD(K,6)
K > 1
N
SUM1=0.0
SUM2=0.0
Y
SUM1 = X+SUM1
SUM2 = MISMOD(K,6)+SUM2
CONTINUE
MISCAP = 10.0*(1.0+(SUM1/SUM2))
ENGTYP > 1
Y
ENGTYP ≥ 2
Y
N
N
STAGLEN = 7.31733*
THRUST**(-0.02793)
STAGLEN = 6.00296*
THRUST**(-0.00958)
STAGLEN = 6.3699*
THRUST**(-0.02156)
10

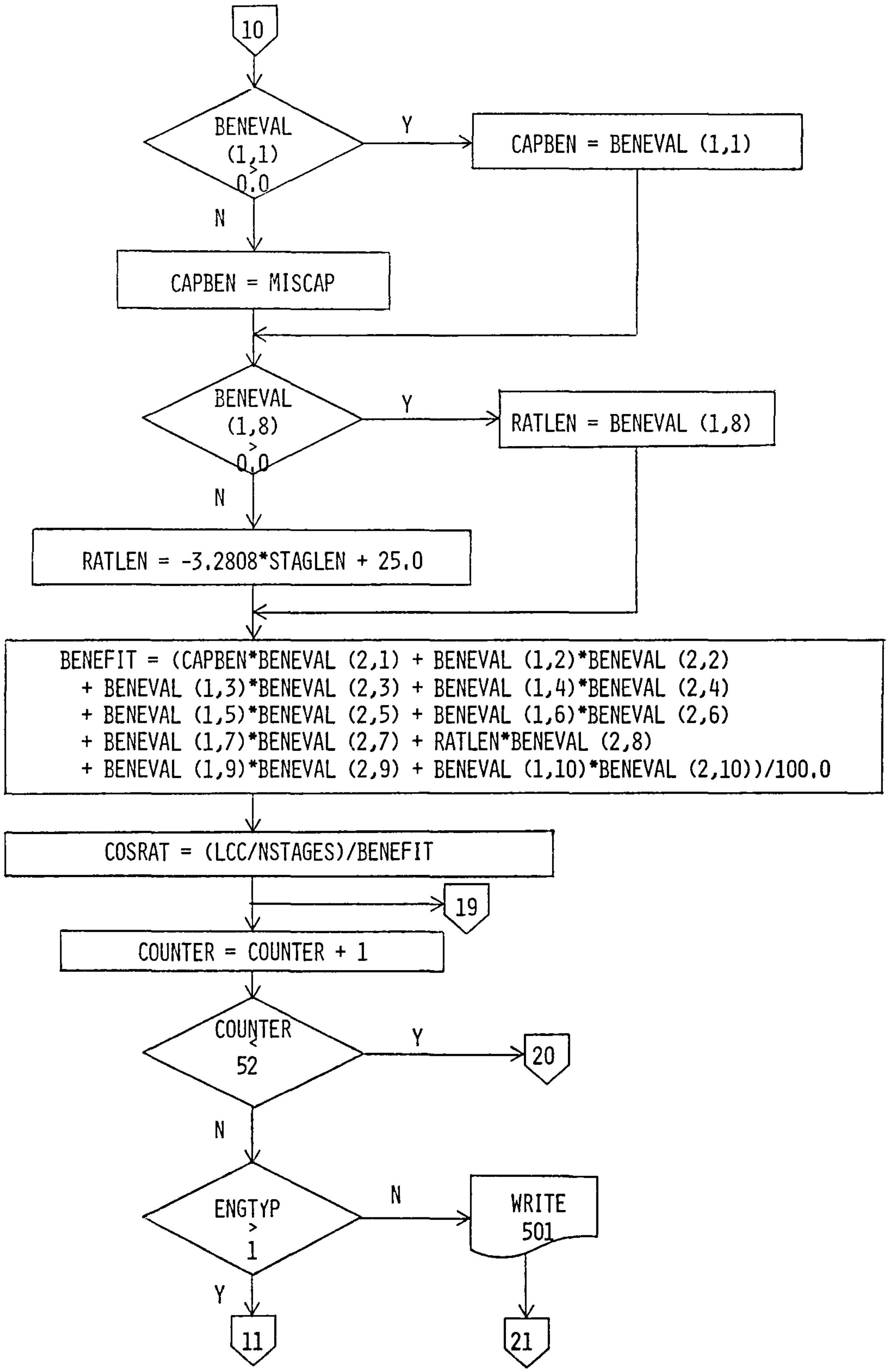

10
BENEVAL (1,1) > 0.0
Y
CAPBEN = BENEVAL (1,1)
N
CAPBEN = MISCAP
BENEVAL (1,8) > 0.0
Y
RATLEN = BENEVAL (1,8)
N
RATLEN = -3.2808*STAGLEN + 25.0
BENEFIT = (CAPBEN*BENEVAL (2,1) + BENEVAL (1,2)*BENEVAL (2,2)
+ BENEVAL (1,3)*BENEVAL (2,3) + BENEVAL (1,4)*BENEVAL (2,4)
+ BENEVAL (1,5)*BENEVAL (2,5) + BENEVAL (1,6)*BENEVAL (2,6)
+ BENEVAL (1,7)*BENEVAL (2,7) + RATLEN*BENEVAL (2,8)
+ BENEVAL (1,9)*BENEVAL (2,9) + BENEVAL (1,10)*BENEVAL (2,10))/100.0
COSRAT = (LCC/NSTAGES)/BENEFIT
19
COUNTER = COUNTER + 1
COUNTER < 52
Y
20
N
ENGTYP > 1
N
WRITE 501
Y
11
21

TABLE B-4 RACE FLOWCHART (CONT'D)

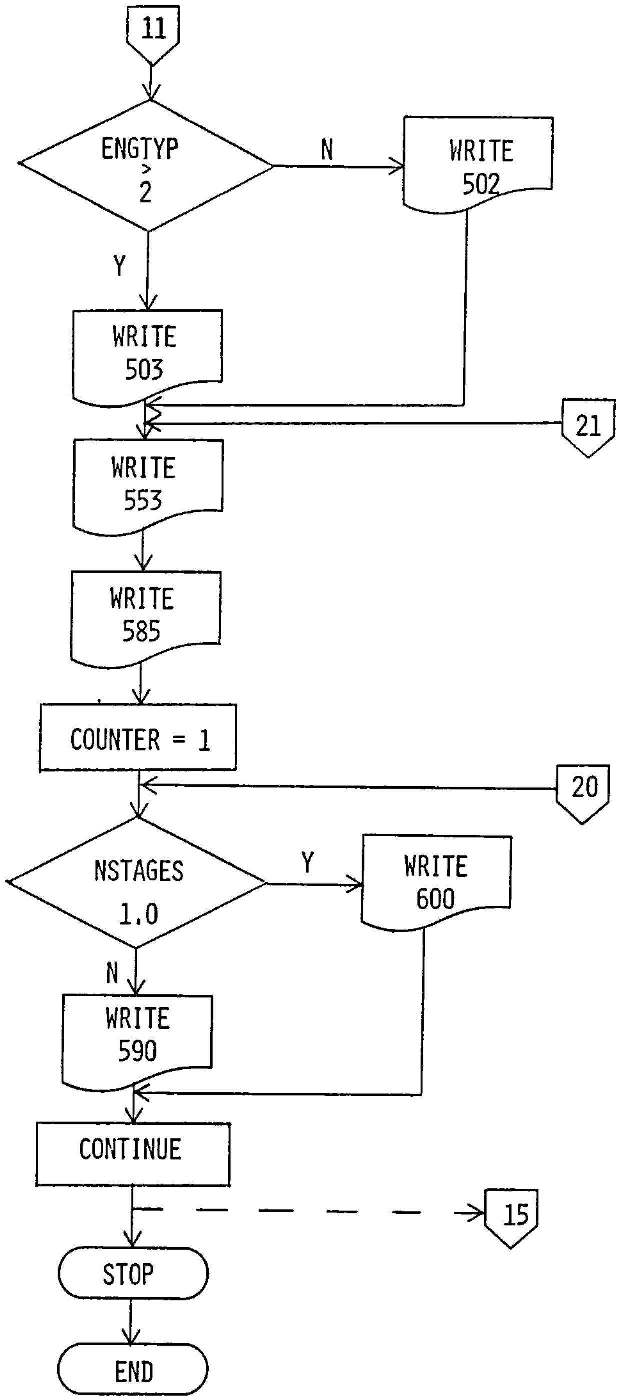


TABLE B-5 SAMPLE RACE OUTPUT

COST/BENEFITS OF ADVANCED ENGINE

INITIAL THRUST= 4450 NEWTONS FINAL THRUST= 10000 NEWTONS THRUST INCREMENT= 50 NEWTONS

MISSION INFORMATION

MISSION NUMBER	PAYLOAD MASS (KG)	MOST CONSERVATIVE ACCELERATION (G)	LEAST CONSERVATIVE ACCELERATION (G)	PROBABILITY	NUMBER OF STAGES	TYPE OF STAGE
1.0	2400 0	999	1.001	1 00	1 0	1 0
2 0	1600 0	020	050	95	1 0	1 0
3 0	4540 0	200	1 000	1 00	1 0	2 0
4 0	4550 0	020	100	1 00	8 0	1 0
5.0	3090 0	.100	200	50	2 0	1 0
6 0	4100 0	100	500	1 00	12 0	2 0
7 0	5900 0	.050	200	20	2 0	1 0
8 0	4540 0	150	400	10	2 0	2 0
9 0	3030 0	050	200	.85	2 0	2 0
10 0	6800 0	010	100	20	8 0	2 0
11 0	3100 0	010	100	1 00	16 0	1 0
12 0	7500.0	050	200	70	4 0	1 0
13 0	7260.0	249	.251	80	1 0	2 0
14 0	8200 0	050	350	50	4 0	1 0
15 0	8200 0	050	350	30	2 0	1 0
16 0	7260 0	050	350	50	2 0	2 0

BENEFIT INPUT MATRIX

	MISSION CAPTURE	RELIABILITY ENGINE	TECHNICAL RISK	GROWTH POTENTIAL	LENGTH OF DEVELOPMENT
RATING	0.00	9 30	3 00	4 00	5 00
WEIGHTING FACTOR	65.0	10 0	5 0	5 0	5 0

	TECHNICAL DESIRABILITY	RELIABILITY STAGE	STAGE LENGTH	FABRICABILITY	REPAIRABILITY (IN ORBIT)
RATING	5.00	5.00	0 00	4 00	0 00
WEIGHTING FACTOR	0 0	10.0	0 0	0 0	0 0

TABLE B-5 SAMPLE RACE OUTPUT (CONT'D)

PROPULSION SUBSYSTEM MASSES (KG)

TELEMETRY, TRACKING, AND COMMAND MASS	ATTITUDE CONTROL MASS	ELECTRICAL POWER SUPPLY (KG X WATTS)	PROPELLANT TANK MASS LH2	PROPELLANT TANK MASS LO2	STRUCTURE MASS	FEED AND DUMP SYSTEM MASS	PRESSURIZATION SYSTEM MASS	PASSIVE THERMAL CONTROL MASS
160 0	200.0	360000 0	170 0	90 0	505.0	195 0	209 0	258 0

MISSION MODEL MATRIX

MISSION NUMBER	MOST CONSERVATIVE THRUST,N	LEAST CONSERVATIVE THRUST,N	MIN. THRUST TO DELIVER PAYLOAD, N	MISSION CAPTURE INDEX SLOPE	MISSION CAPTURE INDEX INTERCEPT
1 0	50827.4	50928.8	0	- 986E-02	502.1654
2.0	874 5	2178 5	0	- 767E-03	1.6707
3.0	14417.2	71813 2	6	- 174E-04	1 2512
4.0	1451 6	7230 2	6	- 173E-03	1.2512
5.0	5801 8	11579 8	0	- 173E-03	2 0041
6 0	6789 9	33811 9	1	-.370E-04	1 2513
7 0	4281 8	17079 7	99 3	-.781E-04	1.3346
8.0	10820 2	28787 1	6	- 557E-04	1.6022
9.0	2877.5	11462 4	.0	- 116E-03	1 3352
10 0	947 3	9432 9	1606 4	- 118E-03	1 1116
11.0	585.1	5811 5	0	- 191E-03	1 1120
12 0	5065 1	20213 1	10970 4	- 660E-04	1 3344
13 0	24570.8	24767.8	5797.9	- 508E-02	125.7160
14 0	5407.9	37739.2	63106.9	- 309E-04	1 1673
15 0	5407 9	37739.2	63106.9	- 309E-04	1 1673
16 0	4947 6	34517.0	5797 9	- 338E-04	1 1673

ALL COSTS ARE IN MILLIONS OF 1982 FISCAL YEAR DOLLARS

LIQUID ROCKET ENGINE RDT&E COST= 2 70E+02 LIQUID ROCKET ENGINE FIRST UNIT COST= 2.80E+00

STAGE WITHOUT ENGINE TOTAL RDT AND E COST= 198 579 STAGE WITHOUT ENGINE FIRST UNIT COST= 22 159

PRIMARY PROPULSION SYSTEM RDT&E COST = 468 579 PRIMARY PROPULSION SYSTEM FIRST UNIT COST = 24 959

TABLE B-5 SAMPLE RACE OUTPUT (CONCL)

COST/BENEFITS OF ADVANCED ENGINE

ALL COSTS ARE IN MILLIONS OF 1982 FISCAL YEAR DOLLARS

THRUST (NEWTONS)	MISSION CAPTURE RATING	PRODUCTION COST	NUMBER OF STAGES	AVERAGE UNIT COST	LAUNCH COST	DEPLOYMENT COST	SUPPORT COST	BENEFIT	LCC	LCC PER STAGE PER BENEFIT
4450	6 307	866 257	54 0	16 042	2300 200	5 512	2305 712	6 130	3640 548	10 998
4500	6 272	866 257	54 0	16 042	2300 200	5 512	2305 712	6 107	3640 548	11 040
4550	6 236	866.257	54.0	16.042	2300 200	5 512	2305 712	6 083	3640 548	11 082
4600	6 200	866 257	54 0	16.042	2300 200	5 512	2305 712	6 060	3640 547	11 125
4650	6 164	866.257	54 0	16 042	2300 200	5 512	2305 712	6 037	3640 547	11 168
4700	6 129	866 257	54 0	16 042	2300 200	5 512	2305 712	6 014	3640 547	11 211
4750	6 093	866 257	54.0	16.042	2300 200	5 512	2305 712	5 990	3640 547	11 254
4800	6 057	866 257	54 0	16 042	2300 200	5 512	2305 712	5 967	3640 547	11 298
4850	6 021	866 257	54 0	16 042	2300 200	5 512	2305 712	5 944	3640 547	11 342
4900	5 986	866 257	54 0	16.042	2300 200	5 511	2305 711	5 921	3640 547	11 387
4950	5 950	866 257	54 0	16 042	2300 200	5 511	2305 711	5 897	3640 547	11 432
5000	5 914	866 257	54 0	16.042	2300 200	5 511	2305 711	5 874	3640 547	11 477
5050	5 878	866 257	54 0	16 042	2300 200	5 511	2305 711	5 851	3640 547	11 523
5100	5 842	866 257	54 0	16.042	2300 200	5 511	2305 711	5 828	3640 546	11 569
5150	5 807	866 257	54.0	16.042	2300 200	5 511	2305 711	5 804	3640 546	11 615
5200	5 771	866 257	54 0	16.042	2300 200	5 511	2305 711	5 781	3640 546	11 662
5250	5 735	866 257	54 0	16 042	2300 200	5 511	2305 711	5 758	3640 546	11 709
5300	5 699	866 257	54 0	16.042	2300 200	5 511	2305 711	5 735	3640 546	11 756
5350	5 664	866.257	54 0	16.042	2300 200	5 511	2305 711	5 711	3640 546	11 804
5400	5 628	866 257	54 0	16 042	2300 200	5 511	2305 711	5 688	3640 546	11 852
5450	5.592	866 257	54 0	16.042	2300 200	5 510	2305 710	5 665	3640 546	11 901
5500	5 556	866.257	54 0	16 042	2300 200	5 510	2305 710	5 642	3640 546	11 950
5550	5 521	866 257	54 0	16.042	2300 200	5 510	2305 710	5 618	3640 546	11 999
5600	5 485	866 257	54 0	16.042	2300 200	5 510	2305 710	5 595	3640 545	12 049
5650	5 449	866 257	54 0	16 042	2300 200	5 510	2305 710	5 572	3640 545	12 100
5700	5 413	866 257	54 0	16 042	2300 200	5 510	2305 710	5 549	3640 545	12 150
5750	5 378	866 257	54 0	16 042	2300.200	5 510	2305 710	5 525	3640 545	12 201
5800	5 602	907 486	57 0	15.921	2467 300	5 816	2473 116	5 671	3849 181	11 907
5850	5.582	683 380	41.0	16 668	1966 500	4 184	1970.684	5 659	3122 643	13 460
5900	5 568	683 380	41 0	16.668	1966 500	4 184	1970 684	5 649	3122 643	13 483
5950	5 553	683 380	41 0	16 668	1966 500	4 184	1970 684	5 639	3122 643	13 506
6000	5 538	683 380	41 0	16 668	1966 500	4.184	1970 684	5 630	3122 642	13 529
6050	5 523	683 380	41 0	16 668	1966 500	4 184	1970 684	5 620	3122 642	13 552
6100	5.508	683 380	41.0	16 668	1966 500	4 184	1970 684	5 611	3122 642	13 575
6150	5.494	683 380	41 0	16 668	1966 500	4 184	1970 684	5 601	3122 642	13 598
6200	5 479	683 380	41 0	16 668	1966 500	4 183	1970 683	5 591	3122 642	13 622
6250	5 464	683 380	41 0	16 668	1966.500	4 183	1970 683	5 582	3122 642	13 645
6300	5.449	683 380	41 0	16 668	1966 500	4 183	1970 683	5 572	3122 642	13 668
6350	5.435	683.380	41 0	16.668	1966 500	4 183	1970 683	5 562	3122 642	13 692
6400	5 420	683.380	41 0	16 668	1966 500	4 183	1970 683	5 553	3122 642	13 716

Thrust level iterations, costs, and benefits are the results shown on the last page of Table B-5. The parameters of primary interest are Benefit, LCC (Life Cycle Cost), and LCC per captured stage per benefit point. The fourth column, number of stages, implies the number of stages capture compared to the total number of stages in the mission model. A mission is considered "captured" if $\Omega > 0.0$, thus an additional integer number of stages appear in this column.

First-order effects of "number of stages" present themselves in production cost, launch cost, and deployment cost. As more missions are captured, the costs increases as expected.

LCC is composed of production cost, support cost, and PPS RDT and E cost. The driver of LCC is support cost ($\approx$ 65%), followed by production cost ($\approx$ 25%), and least influential of the three, RDT and E cost, ($\approx$ 10%).

End of Document